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Design, Simulation, Management and Control of a Cooperative, Distributed, Earth-Observation Satellite System

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If I had a world of my own, everything would be nonsense.
Nothing would be what it is, because everything would be what it isn't.
And contrary wise, what is, it wouldn't be.
And what it wouldn't be, it would.
You see?

Alice's Adventures in Wonderland
Lewis Carroll, 1865

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CHAPTER 1

Introduction

The research presented in this thesis explores the potential of, and develops a framework for, the application of fractionated satellite systems to science-dedicated Earth observation missions. The majority of satellites are actually designed according to a *monolithic* (or conventional) architecture; this simply means that a satellite is an autonomous machine, able to accomplish its mission and to provide by itself to its needs (power, communications, attitude control) and only requires a minimum, periodical, human supervision. This approach has proven to be extremely functional and successful, as the last sixty years of space exploration demonstrate. However, over the years, some non-optimal features of the conventional satellites have been highlighted: they tend to be single-designed, thus leading to expensive re-design every time a new mission is proposed, subject to performances degradation and instruments ageing, with minimum-to-none components standardisation, that result (among the others) in time-consuming integration and validation processes and limited End Of Life (EOL) possibilities apart from decommission. Furthermore large satellites are more likely to exceed initial allocated financial and temporal budgets, thus possibly leading to the cancellation of the whole project in spite of a part of the spacecraft has already been developed and build.
Several solutions have been proposed to address these undesirable characteristics: an innovative approach proposes to physically separate the subsystems into different modules exploiting a wireless interconnection [BE06b]. A module itself is a single-task designed satellite and can be added, or exchanged, independently from the others, as well as be reused over different missions, Fig. 1.1. Such an architecture is labelled as *fractioned satellites* [BE06a] to highlight the physical distribution of functionalities (e.g. power generation, telecommunication, etc.) over a cluster of orbiting elements.

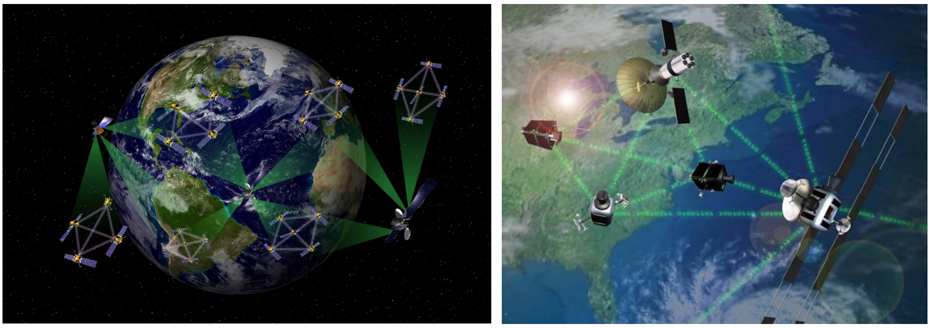


Figure 1.1: F6 fractionated satellite concept, Courtesy of DARPA

The resultant distributed system can be seen as a free-flying payload supported by free-flying service modules. In general, the paradigm shift towards using a multiple-satellite cluster has also been fuelled by the perceived advantages of increased robustness, greater flexibility, and in order to accomplish the large-scale geometries imposed by specific science objectives [WHMK12, GWHR12]. Small distributed spacecraft could also guarantee better coverage than monolithic with almost comparable performances due to sensors miniaturisation [TK12].
There are many ways to implement the fractionation [MW06]. By interpreting literally the idea, it is possible to de-couple entirely the subsystems using different modules thus creating a completely heterogeneous system. In order to cut down the costs it seems reasonable to produce standard buses for every subsystems. Nonetheless a complete functional decomposition with the current technology not only is impractical, it is physically impossible. Every module must be able to provide by itself to basic functions like power distribution or thermal control as well as structural integrity. Thus the most logic configuration for a fractionated spacecraft is a combination of shared resources and module-owned properties [MW06, LC08].
Apart from design issue, operation phase poses a new class of challenges by itself: satellites have been considered and designed as stand-alone elements. No cooperation among different satellites was needed and the ongoing missions involving several satellites working on a single project are a composition of self-standing spacecraft that observe the same phenomenon using dissimilar instruments or from a diverse perspective. The Cluster mission [RCWS93] and the A-Train [PWK06] are good examples of this philosophy: in spite of being composed of multiple (and even heterogeneous) satellites they do not require interactions among them, basically they only work on the same topic. The introduction of the fractionated approach requires a new methodology to control and coordinate the spacecraft system's efforts in order to guarantee that remote resources will be gathered and distributed according to the satellites needs; furthermore the proposed concept has been thought to be scalable to large systems, possibly involving tenth of different elements. The operational costs of monitoring and commanding a large fleet of close-orbiting satellites is likely to

be unreasonable unless the on-board software is sufficiently autonomous, robust, and re-configurable [Mue06]. The first step in inter-satellite cooperation has been made almost two decades ago, with the commissioning of the first Tracking and Data Relay Satellite (TDRS) [Wea12]; working as remote data storage and re-transmission hub, it could be interpreted as a form of fractionation of the communication system, that lead to augmented performances of the connected elements due to the downlinks' larger bandwidth and duration compared with the traditional Space-Earth connections. However fractionation not only offers a possible increment is satellite capacities, but it requires new methods to address the design and test the foreseen performances; the challenge for space engineers is to find the way to obtain the advantages that such a philosophy involves.
The aim of this work is to develop a methodology to optimise the performances of a series of wireless connected spacecraft is order to compare them with a traditional design satellite.

1.1 Fractionated Satellites

Satellites are complex and expensive systems envisioned to operate in a hostile environment that reduces their reliability over time. Due to the challenges they must face, their design is limited to the essential requirements provide by the mission payloads. Many methods have been offered for enhancing spacecraft flexibility, including on-orbit satellite servicing [Lon05, Rey99, JH06], staged constellation deployment [ddC03], and on-orbit software upgrades [Nil05]. Each of these methods involves one or more value-enhancing attributes, that could be collectively called as design flexibility: this can take the form of capability restoration, capability augmentation, risk diversification, schedule diversification, or uncoupling of system requirements. A proposed architectural approach that could enhance the lifecycle value of a spacecraft through flexibility is the adoption of a fractionated satellite architecture [BE06b].
A fractionated architecture is one in which the spacecraft system is decomposed into multiple modules which interact wirelessly to deliver at least the same capability as that provided by a comparable traditional, monolithic system [BE06a]. There are fundamentally 2 possible approaches to fractionation: heterogeneous and homogeneous. The former foresees that the original spacecraft is decomposed into functionally dissimilar modules. For instance, a spacecraft with a separate payload, tracking telemetry and communications, and computation and data handling modules would be considered to be fractionated into three heterogeneous modules, Fig. 1.2. The latter, homogeneous fractionation, is applied when a spacecraft is decomposed into a number of identical modules. A possible example of this could be the constellation originally designed for the Terrestrial Planet Finder mission [HLA$^+$04] with multiple identical sensing satellites serving as a distributed aperture in space. With either type of fractionation, one of the critical driving factors is the level of connectivity

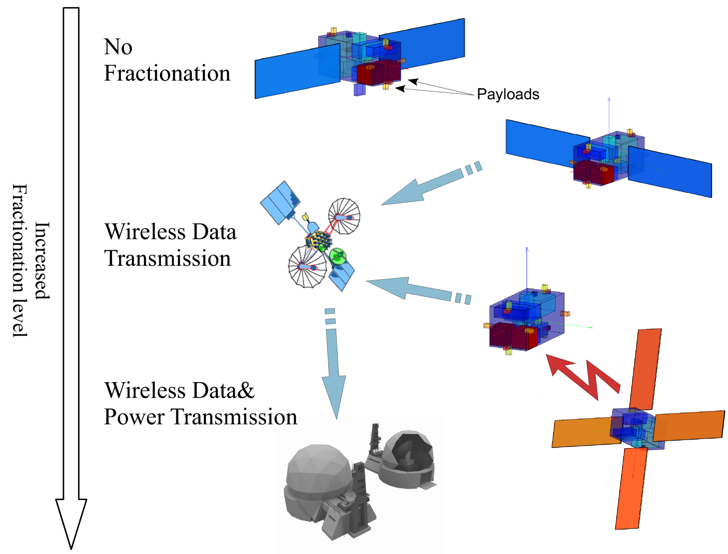


Figure 1.2: Heterogeneous fractionation

between the separate modules: what resources have to be distributed, how, to what extend are questions that must be answered during the preliminary design phase in order to guarantee that the resulting system will behave consistently. However, there is no rule-of-thumb to decide the fractionation level or how to implement it, thus the development of a metric to evaluate the performances as a function of the adopted configuration is required.
There are significant possible advantages related to this kind of design [BE06b]:

- Diversification of launch and on-orbit failure risk
- Reliability enhancement through emergent sharing of subsystem resources
- Scalability in response to service demand fluctuations
- Upgrade-ability in response to technological obsolescence
- Incremental deployment of capability to orbit
- Graceful degradation of capability on-orbit
- Robustness in response to funding fluctuations and requirements changes
- Reduced integration and testing due to subsystem decoupling
- Production learning across multiple similar modules
- Enabling spacecraft to be launched on smaller launch vehicles with shorter time-scales
- Requirements diversification

By requirements diversification is here intended that if service subsystems (or at least most of them) are removed from the module that carries the payload, this could be designed and operated according to payload functions only -i.e. attitude then would be non longer constrained by solar panels, thermal radiator or antenna orientation-. The paradigm is largely based on last years advances in wireless and network-related technologies, mainly developed for terrestrial applications, as well as the increased diffusion of micro-satellites (that already exploit some of the listed advantages) [VBM+11]. Technologies like self-forming networks [KSP99], secure wireless communication [YNK01], distributed computing [FKT01] can be considered ripe and ready for deployment. Contrary-wise, other aspects of fractionation are not so well established: as en example, efficient wireless power transfer, although several proof-of-concept exist, is still in development phase.
This highlights one of the main disadvantage of the proposed architecture: some of the involved technologies are not yet fully mature and therefore their exploit requires further research and development. Furthermore fractionation can neither fully replace the on-board subsystems nor it can be applied to every subsystems; minimum as well as non-distributive functionalities must be guaranteed on all modules, leading to the duplication of some functions and hardware. As a result, the initial cost of such a system is expected to be higher than that of a monolith, whereas the lifecycle cost is expected to be lower [BE06b, BE06a].

1.1.1 System F6 program

The Future, Fast, Flexible, Fractionated Free-flying Spacecraft United by Information Exchange or F6 program was a DARPA funded initiative to investigate and realise a spacecraft system with a fractionated architecture [rep10]. It was predicated on the development of open interface standards- from the physical wireless link layer, through the network protocol stack, and including the real-time resource sharing middle-ware and cluster flight logic-that can enable the emergence of a space "global commons" which would enhance the mutual security posture of all participants through interdependence.
A key program goal was the industry-wide promulgation of these open interface standards for the sustainment and development of future fractionated systems. The program would have culminated with an on-orbit demonstration in mid 2015 of the key functional attributes of fractionated architectures. The on-orbit demonstration would have taken place in LEO, and would have been approximately six months in duration, with a potential subsequent residual capability demonstration lasting up to 18 months. The successful completion of these on-orbit demonstrations constitute the high-level objectives of the program, from which proposers and performers were expected to derive system-, subsystem-, and component-level objectives:

- Capability for semi-autonomous long-duration maintenance of a cluster and cluster network, and to add and remove spacecraft modules to/from the cluster and cluster network.

- Capability to securely share resources across the cluster network with real time guarantees and among payloads or users in multiple security domains.
- Capability to autonomously reconfigure the cluster to retain safety- and mission critical functionalities in the face of network degradation or component failures.
- Capability to perform a defensive cluster scatter and re-gather manoeuvre to rapidly evade a debris-like threat; the planning and execution of the scatter and re-gather manoeuvres shall be performed without intervention or communication from ground operations.

In 2008, DARPA announced that contracts for the preliminary development phase of the System F6 program were issued to teams headed by Boeing, Lockheed Martin, Northrop Grumman, and Orbital Sciences. The second phase of the program was awarded to Orbital Sciences, along with IBM and JPL, in December 2009 mission but later terminated that deal. The agency restructured the program, distributing work among several small companies and universities, with none assigned the lead integrator role. The lack of rationale behind the leaderless contracting structure, software development delays and contractor performance issues, and finally the fact that System F6 demonstration did not have a traditional mission -such as imaging-resulted in program cancellation on May 17, 2013 [Spaa].

1.2 Automated Satellite Design

Several systems engineering tools exist for space systems design. Most of them have been developed by different organisations with different uses in mind. Tab. 1.1 lists some of the major systems engineering tools in use today for space systems design (several others already exist or are under developed at the time of writing). Private companies (The Aerospace Corporation, Ball Aerospace), universities and space agencies have created tools oriented according to their main area of interest (e.g. planetary exploration missions for NASA and ESA, commercial, communication satellites for stakeholders companies) thus leading not only to the realisation of software but in some cases dedicated concurrent design facilities to provide computer-aided analysis, design, and trade studies for the missions [SBMF10]. It is worthy to highlight that apart from the Multiobjective, Multidisciplinary Design Optimization Systems Architecting Methodology (MMDOSA), the other tools are focused on the design of a single spacecraft rather than an entire distributed satellite system.
According to literature review, only one fractionated satellite system engineering tool have been developed so far [Tat12]; its authors have integrated the fractionated concept within an existing satellite design optimisation methodology.

The implementation of a new tool instead of the modification of a pre-existing one has been preferred considering:

Name	Organisation	Use
COBRA	The Aerospace Corporation	Automated assessment of program cost risk and schedule risk as a function of spacecraft complexity for interplanetary missions [Bea00]
CEA	The Aerospace Corporation	Mapping of "what if" cost and performance trade studies for Air Force missions [WL96]
ESSAM	Univ. of Colorado	Small Satellite bus component selection [Rid98]
GENSAT	Computational Technologies	Object-oriented software that interconnects existing commercial satellite subsystem tools and component databases [BPS95]
ICE	CalTech	Concept definition of novel space missions via integrated information systems [SSW98]
MERIT	The Aerospace Corporation	Automated assessment of the cost and performance implications of inserting existing vs. new technologies into a spacecraft bus
MIDAS	JPL	Analysis of proposed spacecraft designs via tool executions on distributed machines [FCM+97]
Modelsat	ROUTES	Cost and mass modelling for comms satellites
PTM	JPL	Cost and performance prediction of novel interplanetary and space science missions [Bri95]
QUICK	JPL	Spacecraft design programming language with extensive component databases and scaling relationships for spacecraft design [Ski92]
SCOUT	The Aerospace Corporation	Single spacecraft mission bus component and launch vehicle selection [Mos98]
SMALLSAT	NASA Langley Research Center	Earth observation spacecraft sensor and satellite bus configuration [Gre92]
SpaSat	Ball Aerospace	A preliminary spacecraft sizing, cost estimating orbital analysis tool
MMDOSA	MIT	Multiobjective MDO for the conceptual design of distributed satellite systems [Jil02]
SPIDR	USC	An artificial intelligence-based search and optimization engine for conventional and fractionated satellite systems [Tat12]

Table 1.1: Space Systems Engineering Tools

- availability: most of the listed software are commercial or free-to-use for research and academics for only USA-based organisations.
- customizability: fractionation requirements are radically different from traditional satellites, thus requiring an open-source tool to perform all the necessary modifications.
- close-the-loop necessity: the aim of the work is to design and test the satellite system, then outputs in the form of metrics and costs will not suffice to this end -the whole spacecraft design, including the component list are required-

1.3 Satellite Simulation

By definition, *simulation* is an imitative representation of a real-world system that allows its operator to examine functions, attributes, and behaviours that may not be practical to deal with using direct analysis or experimentation. Given a set of inputs, a simulation uses a model(s) to provide a set of outputs. Simulations are commonly used to deal quantitatively with large, complex systems for which there does not exist an analytic system of equations with a closed-form solution [SC95]. Spacecraft are a perfect example of this kind of systems as they exploit (among the others) non-linear attitude and motion dynamics, time-changing mass properties, non-uniform temperature distribution with time and orientation related boundary conditions. Simulation activities have a vital role to play in supporting operation, research and development both for existing and under study satellite systems. Commonly targeted tasks are:

- Mission definition and demonstration
- Crew training
- Design, prototyping and verification of systems
- Supporting software validation
- Mission/system (end-to-end) modelling
- Attitude and Orbit Control System, Trajectory, Guidance and Control, Navigation simulators
- Mathematical analysis models of payloads systems
- Demonstration and promoting of space systems by real-time visualisation

However, in spite of their importance, satellite simulations has been mostly investigated keeping a certain degree of separation between the functionalities: the dynamical contribute -orbit and attitude- and electro-mechanical part -the subsystems-. The former are analysed in order to evaluate the influence of the perturbations and the effectiveness of the control system, the latter to check specific parameters under controlled (critical) conditions.
As a result, structure and granularity of models required by subsystems vary greatly

compared to those developed for simulation and control. Mirroring, and causing in part the wide gap between techniques under research at labs and the ones being utilised on operational spacecraft, is the distance between the fidelity and representativeness of models in use at research centres and the proprietary simulations developed in industry [IATM+12]. Satellites, compared to other types of machinery, have a tighter bond with their environment: its influence determines the capacity to generate power, the temperature, the possibility of communicate and operate. To exacerbate the problem, boundary conditions affect the spacecraft regardless for functional distinctions that have been made by designers: simulators dedicated to the power generation take the solar panel temperature as a given parameter, whereas it is a function of the orbit, attitude and dissipated power [LCKL88]. Similarly thermal analysts base their work on given attitude and power load under nominal conditions. When subsystems are analysed singularly, there is a loss of information due to the non considered cross-influences among them. In order to investigate these aspects the commonly used aggregate models, such as transfer functions and state space descriptions, have limited applications and multi-system models of component behaviour are required. In order to effectively introduce the fractionated concept in the simulation framework, a specific focus on subsystem performances and available resources is required: the satellite system will operate under shared resource conditions, thus their evaluation as a function of time, attitude, operative conditions is as important as the relative position and orientation of the spacecraft. Furthermore these analysis must include the mutual influences among the subsystems. There is a

Name	Organisation	Orbit	ADCS	Subsystems	Distributed
STK	Agi	✓	✓	✓	?
Open-SESSAME	Virginia Tech	✓	✓	?	X
G-SDSU	NASA	✓	✓	?	?
SIMSAT	ESA	✓	✓	?	✓
VSRF	ESA	✓	✓	✓	✓

Table 1.2: Spacecraft Systems Simulation Tools

large number of available spacecraft simulators, mainly dedicated to orbital and attitude aspects [Sch04]; systems can be modelled using external tools (as Finite Element Solvers for thermal and structural models) and subsequently integrated once a general framework has been created. Tab. 1.2 reports some possible examples of simulation tools that have been developed for commercial and research applications that already exploit these functionalities. None of them allow the execution of the tested and validated sub-system (e.g. the AOCS control loop) “in context”, together with simulated equipment (e.g. thermal, power, communications) in a dedicated simulation environment. STK has a commercial licence that includes data management, sensors and Telemetry Tracking and Command (TT&C) analysis but has limited power and thermal capacities that could be integrated using external tools via plug-ins[Ana11].

Open-SESSAME was stated as a master thesis project and then further developed [TH03]; distributed with a GNU licence is mainly dedicated to orbital and attitude aspects with related communication evaluation possibilities. The Goddard Satellite Data Simulator Unit (G-SDSU) is a payload-oriented tool that enables users to insert their own satellite simulator to convert model-simulated atmosphere states to various types of satellite observable signals [NASb]. SIMSAT is a general-purpose real-time simulation infrastructure developed for the European Space Agency; it comes with a variety of semi-standard models for ground modelling, environment mode ling etc., within a reference architecture [Whi07]. ESA's The Virtual Spacecraft Reference Facility is a simulation facility where under proposed satellites can be extensively tested, including subsystem, components, on-board software with or without hardware in the loop; however, being a facility and not just a tool its exploitation within the contest of this work could not be possible[ESAb]. None of the mentioned projects in its off-the-shelf version satisfies the distribute satellites interaction and subsystems models requirements that are required in order to be able to simulate the operation of a fractionated satellite system. The required extensive modifications could be implemented on the open-source software only, reducing the range of the possible choices. Due to similar consideration done for the available engineering tools and the time required to introduce the required adjustments, the development of a dedicated simulation tool has been preferred.

CHAPTER 2

Thesis Overview

The goal of this research is to develop a methodology to design and simulate distributed satellite systems in order to evaluate quantitatively how fractionation affects not only the cost of the system, but also its performances once deployed. This aim requires both preliminary design and operation phase to be analysed: no satellites with fractioned architecture actually exist, so there is not a benchmark that could be used as reference.
The simulation phase could be implemented only when a candidate spacecraft is provided; and in order to design a satellite with such a peculiar concept, extensive modifications to current design methods are required. Furthermore the unique features introduced by the resource sharing are beyond tradition spacecraft operations, thus needing an ad hoc tool to consider and test them. In order to compare the performances of the fractionated system with an analogous monolithic satellite, objective quantities have to be evaluated, like the total cost of the spacecraft including estimated development and research, ground support, construction, integration and launch. The specific objectives of the proposed research follow:

- To develop a framework to address the design of a distributed and cooperative satellite system with shared resources.
- To analyse from the operative point of view the implication of the fractioned approach.
- To develop a methodology to evaluate the performances of the created satellite.

In order to achieve the specified objectives, the thesis could have been divided, both from temporal and logic points of view, into 5 main topics, depicted in Fig. 2.1:

- Satellite design
- Fractioned spacecraft design
- Design optimisation
- Satellite operations
- Multiple satellite operations

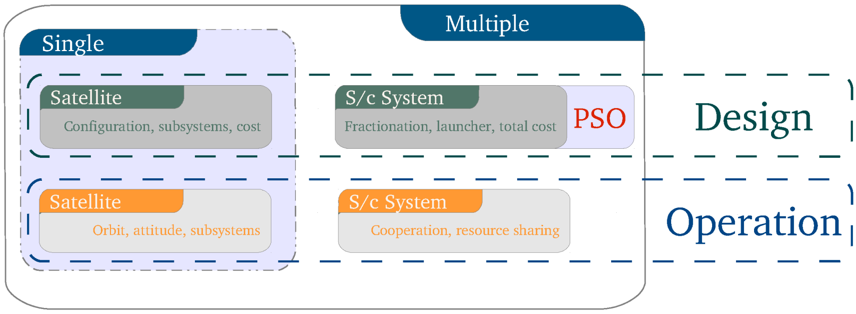


Figure 2.1: Thesis overview

Each of them covers a specific aspect of the work that had to be done in order to understand the real potentialities of the fractionated approach. The task's division has been originated from the study and the analysis of the concept, both using literature, and a priori knowledge of satellite design and operation. The result was the idea that, from a functional perspective, fractionated spacecraft systems could be seen as a more generic approach to satellite deployment, Fig. 2.2-a (whereas from a built-so-far standpoint is exactly the opposite, distributed systems are a small subset of the traditional satellites, Fig 2.2-b). When the whole spacecraft system is considered, fractioned modules possess all the basic capacities of the monolithic elements plus advanced cooperative modes enabled by their additional hardware. According to

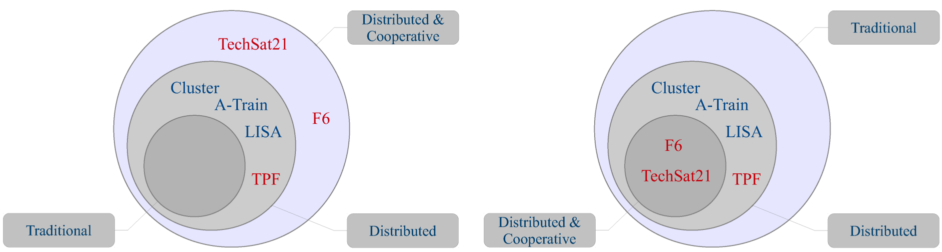


Figure 2.2: Satellite classifications, functional (a) and by number of build elements(b)

these considerations, the logic consequence was to develop a framework that could

analyse a generic spacecraft system; in this sense, a traditional satellite could be seen as a sub-case of distribute/cooperative systems where the number of involved spacecraft is equal to the unity and no resources are shared. But it could also be applied to heterogeneous systems like nowadays science-satellites that rely on the TDRS or similar data-rely spacecraft.
Structure and steps required to full-fill the main objectives are hereby discussed in the following sections.

2.1 Satellite design

As anticipated, the lack of a reference fractionated satellite constitutes a non-negligible problem: without a satellite that could be studied, the development of a metric to estimate its performances would not have any sense. This poses the initial requirement for an engineering tool able to address the preliminary design of a spacecraft, extensively described in Chap. 3. The specific objective of the tool are:

- The capacity to design a science-dedicated LEO satellite with a reasonable degree of confidence
- Provide subsystem power and mass budget, main components list and reliability
- Include the possibility for additional hardware and requirements modification related to resource sharing
- Accept external additional constraints like specific launchers or ground stations
- Sustain and pass a validation campaign using existing satellites as reference

The words “with a reasonable degree of confidence” refer to the fact that such an engineering tool has to face (and solve) within minutes the same problem that an equip of senior engineers and mission technicians would solve in hours or days; their experience and considerations could hardly be translated into a software. However the objective is to obtain a simplified spacecraft that is compatible in terms of mass, power and configuration with the analogous system designed by specialists.

2.2 Fractioned spacecraft design & Design optimisation

This part of the work has been specifically addressed to the impact of fractionation on the spacecraft system design; there are several unknowns related to the resource sharing. Some of the involved technologies, especially for power transfer, are still under test and development and, according to literature, there is a great variability in current and estimated performances. Furthermore as no fractionated spacecraft exist,

there are no data about the influence of the distribute resources on cost or operative features.
In order to design a fractionated spacecraft system with the under analysis configuration, numerous tasks had to be performed:

- Identification of possible shared resources
- Analysis of their current (or near future) TRL
- Study of the necessary additional hardware and satellite requirements modification
- Introduce coherent hardware constrains on master and dependant satellites
- Evaluate the effects on cost and mass parameters due to different type of fractionation (different and amount of resources)
- Use fractions as free variables to optimise the spacecraft system
- Minimise launch vehicle cost according to the new configuration
- Calculate the cost of the entire system

The optimisation is necessary both for design analysis and feasibility of the operation phase: firstly as the whole concept is still on paper, no one really knows how effective it could be (although some literature is available on this point, once again not all of the authors are unanimous [BE06b, BLSE07, O'N10, Spaa]). The exploit of an optimisation tool would allow to evaluate how different fractions affect the spacecraft in combination with the original requirements originated by the mission payload.
Secondly improper configurations would emerge during the operation phase, producing meaningless simulations: i.e. if all the satellites have the capacity to communicate with the ground station, they would not need to cooperate among them.
Fractioned spacecraft design will be discussed in Chap. 4. The results achieved including the optimisation loop in the design phase will be presented in Chap. 7, including a brief description of the used method, fitness value calculation and convergence performances.

2.3 Single & Multiple Satellite Operations

The operation phase is aimed at simulate the behaviour of the satellite during its orbit; when design is coherent with the requirements and the spacecraft is operating under nominal conditions, simulations has little utility. However due to the particular features introduced by the fractionation, the operative phase has been modelled to test the additional effects due to having spacecraft that have to access resources that are not located within the vehicle frame and to provide a framework that could be

used in the future to validate cooperation schemes.
Single satellite operation, depicted in Chap. 5 has been the first step taken in this direction.
The simulated model should include:

- Orbit and attitude evolution including effects due to disturbances and controls
- Power subsystem, with evaluations of generated (solar panels), consumed and available (batteries) power
- TT&C subsystems, including long (ground station) and short (other satellites) range connections
- Thermal subsystem, able to evaluate the satellite components temperature and to control them using heaters
- Propulsion subsystem
- Attitude control system with simplified actuator models whose used power and propellant influence Electrical Power System (EPS) and propulsion systems respectively
- GNC algorithm

Considerable attention has been given to subsystems mutual influences, identified through an a priory analysis.
Additional requirements for multiple satellite simulations include:

- Cooperation model for communication
- Relative attitude and position evaluation
- Upgrade of GNC algorithm to manage multiple spacecraft

The development of a distributed heterogeneous simulation infrastructure is also described in Chap. 6, which will serve as a test bed for modelling and simulation activities.

CHAPTER 3

Satellite Design

Spacecraft system design is a multi-disciplinary, labour-intensive, costly, and time consuming process that considers the mission objectives, payload, structure, orbit and attitude dynamics, thermal control, communication, power supply, as well as other parameters [Rid98]. Is typically addressed by gathering subsystem experts that create and update design concepts over the time-scale of days, merging analyses from various heterogeneous modelling tools. Technology could automate and possibly optimise the spacecraft design process to lower costs (and time) while maintaining reliability. Due to the prohibitively expansive search-able space implied in spacecraft design, many tools have been based on limited subset of design variables and empirical formulas [Rid98, ADL98]. The key problem to consider lies in choosing the modelling technique and implementing an optimisation strategy. The following chapter describes how automated satellite has been treated, highlighting limits and made simplifier assumptions.

3.1 Satellite Assembly Procedure

The Satellite Assembly Procedure (SAP) is an iterative algorithm created to address both the preliminary design of conventional satellite and a more specialised cluster of task-dedicated modules, according to the provided requirements; its main structure is represented in Fig. 3.1. The global outputs are the features of each subsystem, the part-list of the main components and the overall properties of the resulting satellite:

- Mass (dry, wet and estimated adapter), Center Of Gravity (COG) position with respect to main structure and inertia tensor

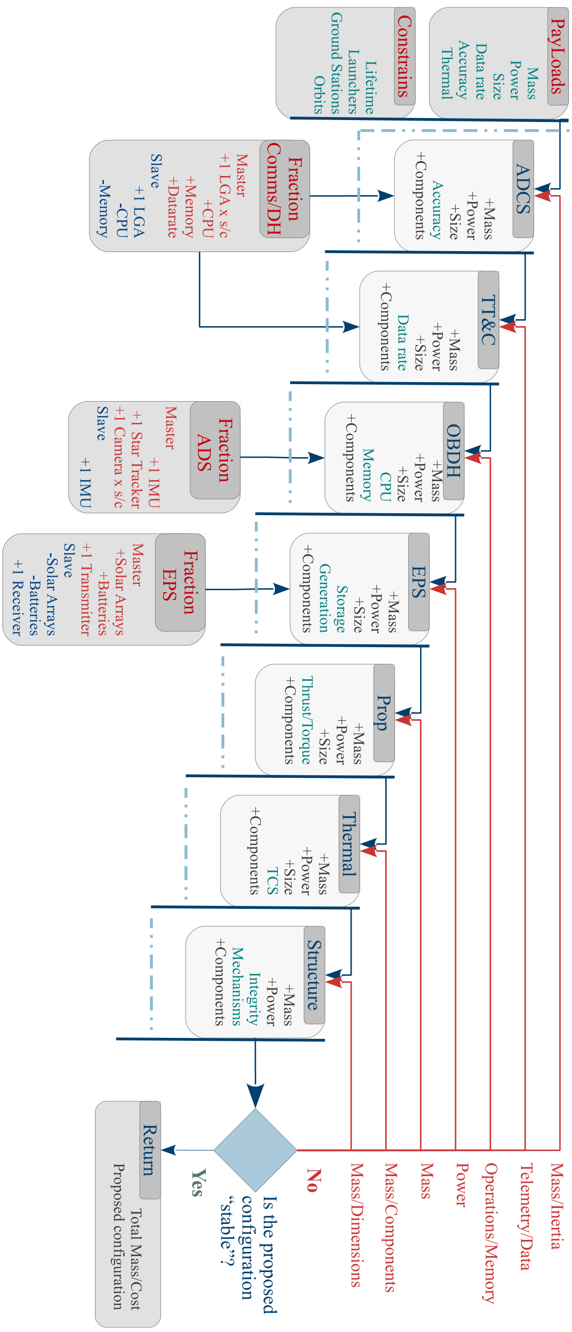


Figure 3.1: Satellite Assembly Procedure

- Costs (research and development, ground supports, launch, for theoretical first unit and following elements)
- Geometry and dimensions (operation phase and folded), impact parameter β, solar array surface
- Power (maximum and minimum, operation and navigation phases), battery capacity, solar panels output
- Downlink/Uplink Data Rate (DR), both science and telemetry
- On-board memory and operations per second
- Systems and satellite reliability

An integrated single/multiple satellite Multidisciplinary Design Optimization (MDO) has been considered but later discarded due to complexity issues highlighted by literature review [CDF+94, Rec91]. Instead, a simpler *bottom-up* approach has been selected: starting from high level requirements and payload features, satellite subsystems are assembled one at a time, updating and using the under-construction solution as revised input. The reasons behind the choice of a less sophisticated and heuristic solver, instead of a more complete one, have been basically time-related; the analysis, design and implementation of a MDO tool would have been hardly compatible with the available time frame. Furthermore the single satellite model is only a part of a larger optimisation process, thus the exploit of a tool able to provide reasonable results in a short time scale (minutes or less) had been of primary interest. The space state of the solutions has been restricted by a priori considerations that have limited the number of possible technologies and approaches to those that are compatible with the science-oriented, Low Earth Orbit (LEO) satellites; these constrains are hereafter reported for every analysed subsystem. At each subsystem level, a set of feasible/compatible solutions is created, and a fitness based on a weighted average of mass, power and cost is calculated for every element in the set. By changing the balance between the weights, mass/power/cost saving solutions can be preferred; unless otherwise specified, only the mass weight is used. The satellite is designed as an assembly of local optimum; the procedure is then repeated as long as stopping criteria are not satisfied. The main criteria is the achievement of a *stable* solution; stability is intended as marginal differences between two consequent iterations ($\pm .1\%$). Parameters tested for convergence are dry and wet mass, maximum and minimum power (both operation and navigation phases), science and telemetry DRs, storage memory. Also configuration, as main components part-list, is checked. A similar mechanic is exploited at subsystem level, thus preventing unnecessary re-design when the input parameters are unchanged. As fail-safe condition, a limit on the maximum number of iterations has been included. Simulations highlighted that a few iterations are enough to produce a stable solution, Fig .3.2; the first iteration is dominated by the payload (and fraction parameters), whereas the second and following iterations are also influenced by the previously constructed subsystems. TT&C, ADCS and EPS

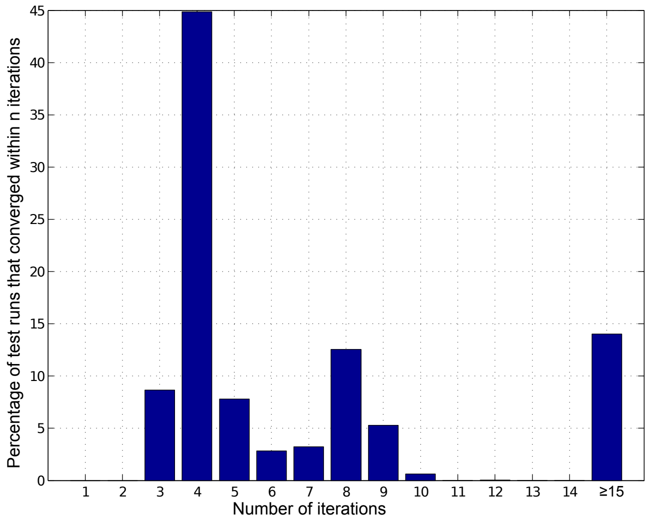


Figure 3.2: Number of iterations before a stable configuration is achieved

usually reach a definite structure within a few iterations, with marginal changes afterwards; structure is the last system to converge (being constructed as continuous and dependant on the other ones). In some cases the algorithm highlighted an oscillatory behaviour (≥ 15 iterations); the reason has been identified in a combination of (typically) two subsystems, each of them with at least two similar feasible solutions and cross-connected input-outputs that cause the procedure to switch back and forth one solution. Even in these cases the complete solution has shown minimal changes in overall parameters. In general, there are no guarantees neither that a solution exists, nor that it is unique.
Algorithm failures arise from:

- Constrains violation at subsystem level (e.g. insufficient reliability, Sec. 3.2, or no components in the available database that match requirements)
- General configuration issues (total mass over available launcher capability)

Furthermore as the design is derived from the sum of locally optimal elements, the complete satellite can be a non-globally optimal solution. A validation campaign has been performed by comparison of actually build satellites with they SAP designed counterparts; used inputs have been orbital features, lifetime and payloads characteristics. Results are reported in Sec. 3.4. A short description of the design and sizing procedure followed for each subsystem is reported.

3.1.1 Inputs

There are 3 classes of input parameters/constrains:

- General constrains: introduced from high-level design choices (partially dictated by the subsequent operation phase).
- Payload features: used to describe quantitatively the payload in terms of dimension, mass, inertia, required power, generated telemetry and scientific DRs, pointing accuracy and thermal limits.
- Fraction parameters: communicate if and how the satellite under construction is connected with other satellites (they will be discussed in Chap. 4).

Considered constrains include mission lifetime, operation (and transfer) orbit parameters, restrictions on available launchers (e.g. EU only) or ground stations (a particular network or a specific station). Those parameters could arise from political and budget concerns instead of technical considerations, thus they have been distinct from engineering issues. Performances reduction over time (solar cell efficiency, battery capacity, thermal coatings) are also considered during design phase.

3.1.2 Payload

The satellite is assembled in order to support and ensure payloads operations, providing structural integrity, attitude pointing/stability, thermal control, power supply, data analysis and download. Payloads parameters are fixed as the SAP is not allowed to modify them. Ensured inputs are:

- Name, type (e.g. telescope, Synthetic Aperture Radar)
- Mechanical features -mass, dimension, inertia, main axis-
- Thermal properties -maximum and minimum survival and operation temperature, dissipated heat, heat capacity, surface absorptivity and emissivity-
- Power requirements -maximum and minimum power during different mission phases-
- Attitude constrains -knowledge, stability, twist rate-
- Science and telemetry DRs during operation phase
- Average duty cycle per orbit
- Design “short-cuts”

Usually payload and satellite design are (at least for some aspects) simultaneous, whereas SAP assumes the payloads as a fixed input; two parameters have been used to counterbalance this lack of cross-influences using information from real satellite configuration to introduce (if necessary) payload autonomous thermal control and on-platform mount. When multiple payloads are provided, mass and power inputs are combined to generate the worst design case (i.e. both payloads active at the same time) whereas the tightest thermal and accuracy requirements are used.

3.1.3 Command and Data Handling

The Command and Data Handling (C&DH) system performs two major functions: it receives, validates, decodes, and distribute commands to other subsystems and gathers, processes, and formats housekeeping and mission data for downlink or use by an on-board computer [LW99]. Due to its functions it is directly influenced by the TT&C subsystem and by the features of the payload. Preliminary sizing of this subsystem is not a trivial task; although some of the required functions can be identified with a reasonable degree of confidence, there is a consistent number of tasks (with consequent computational load, hardware, command output channels, number of stored commands) that require a more detailed design phase in order to be correctly estimated. C&DH used architectures can be generally classified as:

- Single-unit systems, "all-in-one solution", a single unit provides all commands and telemetry functions. However centralised systems installed on large spacecraft bus would require a massive wire harness in order to connect all subsystems and associated interfaces and heath monitors.
- Multiple-unit, distributed systems, on the contrary are made from several physically separated units connected to the subsystems and a single "central"block.
- Integrated systems combine command, telemetry, flight processing and attitude control into one system. A central high-performance processor is in charge of monitoring and controlling simpler, subsystem level, units.

SAP-C&DH module has been largely based on the multiple-unit approach; a central element controls remote units, each in charge for a specific subsystem; multiple reasons are behind this choice: it allows functional decoupling (simplifying the successive operation-phase simulation), divides satellite control from subsystem specific commands execution (thus the C&DH elements sizing can consider only high level functions whereas local boards deal with low-level telemetry gathering and commands execution). In this sense, data handling tasks can be thought as performed by two main functional blocks: data elaboration and storage. In order to size the C&DH, required reliability, number of required operations and storage capacity are considered; two classes of database entries are compared with the requirements, on-board computers, with both process and storage capacities, (N_p elements) and storage units (N_m occurrences). Inputs depend on payload requirements as well as other subsystems commands and telemetry needs (calculated during their own assembly phases). C&DH construction has been resumed within Alg. 1; basically it compares the requirements with the available computer elements. If they are satisfied, the possible solution is saved, otherwise if the problem is a lack in capacity, a couple computer plus additional storage supports solution is evaluated. Reliability is considered, and if needed back-up elements are added to the solution too, as well as estimated harness mass. The possible solutions are evaluated in order to find the best-fitness one.

Algorithm 1 C&DH Assembly

for $i = 1$ to N_p **do** ▷ Cycle over on-board computers
  if $constrain(computer_i|operations, reliability)$ **then**
    if $constrain(computer_i|storage)$ **then**
      Estimate harness
      Solution $(computer_i)$
      Add Solution to $\mathbf{V}$
    else
      for $j = 1$ to N_m **do** ▷ Cycle on storage devices
        if $constrain(computer_i + storage_j|storage, realibility)$ **then**
          Estimate harness
          Solution $(computer_i, storage_j)$
          Add to $\mathbf{V}$
        end if
      end for
    end if
  end if
end for
Search for *best* solution
return C&DH Solution

3.1.4 Telemetry Tracking and Command

The TT&C provides the interface between the spacecraft and the ground systems (or another spacecraft) [LW99]. Commonly it is a two directions communication system, allowing the passage of payload mission data and satellite housekeeping information to the operation centre as well as the reception of users issued commands. Main operations include carrier tracking (mandatory to lock onto an external source/target), command reception, telemetry modulation and transmission, ranging as well as subsystem operations. Sizing process (schematically reported in Alg. 2) is based on requirements and constrains estimated both from high-level prerequisites and subsystem level inputs. *Determine requirements* includes the analysis of orbit, range, coverage, DR, volume and bit error rate; if no user defined constrains are introduced, frequency (S, C, X, Ku and Ka bands) is treated as a free design variable. More likely, restrictions on available Ground Station (GS) and international regulations will restrain the available channels. Components sizing based on link budget [LW99, FSS11] allows to find a solution that minimise a specified cost function (like mass or required power) without violations of requirements and constrains. The TT&C has been divided in three main areas of influence (with a certain abuse of terminology), Fig. 3.3; the transceiver units have been designed to handle mainly telemetry and commands, whereas large-volume science-data download has been addressed with a dedicated transmission line. The two separate channels have different antenna configurations (wide beam, low gain and limited beam, high gain respectively). Suitable elements

Algorithm 2 TT&C Assembly

Determine requirements
while configurations **do**
    Solution()
    Select frequency
    if $designTransceiver\,f(telemetry\,datarate, frequency, power)$ **then**
        Solution += Transceiver/low-gain antenna
    end if
    if $designTransmitter\,f(science\,datarate, frequency, power)$ **then**
        Solution += Transmitter/LNA/high-gain antenna
    end if
end while
Search for *best* solution
return TT&C Solution

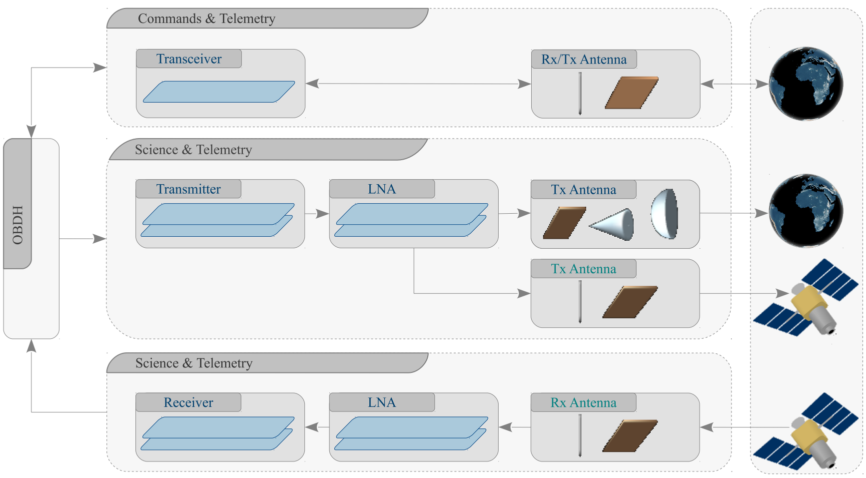


Figure 3.3: TT&C main elements

are selected from a database of off-the-shelf products and then assembled to satisfy reliability and performance levels. When multiple solutions are available (for example several frequencies are available or more than one GS can be used), a candidate that responds at best to the mass, power or cost reduction parameters is selected. Involved components are added to the global list, while mass and power budgets are updated. Additional parts like antenna pointing mechanisms or thermal coatings are provided during the design of the respective subsystems.

3.1.5 Attitude Determination and Control System

The Attitude Determination and Control System (ADCS) stabilises the spacecraft and orients it in the desired directions during the mission in spite of the external disturbances [LW99]. This requires both the capacity to determine the attitude, using sensors, and to control it, by means of actuators. Entry point for subsystem design are mission requirements (type of payload, required accuracy, slew rate), profile and orbital features. These lead to specific ADCS requirements and constrain that restrain the possible subsystem configuration (e.g. pointing accuracy tighter that $0.01°$ can be hardly achieved with horizon sensors and magnetorquers whereas they are a commonly found feature of Star Tracker (ST)/Reaction Wheel (RW) systems). Due to the focus on Earth observation satellites with high pointing accuracy, so far only 3-axis stabilised systems have been considered. A resume of used coupling requirements-technologies can be found in Tab. 3.1. In order to allow regular desaturation manoeuvres, RW and Control Moment Gyro (CMG) must be coupled either with Hydrazine Thruster (hTH), Cold Gas Thruster (cgTH) or Magnetorquer (MT). Number and configuration of the actuators are automatically constrained by their type: RWs and CMGs are placed to form a four-sided pyramid, MTs are disposed orthogonally and parallel to the geometric axes. A total of 12 thrusters is required to ensure a 3 rotational degrees of freedom attitude control. Attitude determination

	Accuracy [°]			Slew Rate [°/sec]	
	$Ac \geq 1.$	$.1 \leq Ac < 1$	$Ac < .1$	$0.05 < SR \leq .5$	$SR > 0.5$
TH	✓			✓	✓
MT	✓			✓	
RW	✓	✓	✓	✓	✓
CMG	✓	✓	✓	✓	✓

Table 3.1: Requirements to attitude control methods

is achieved by using different sets of instruments, mainly ST when high accuracy is required and passive sensors (magnetometers, sun sensors, horizon sensors) otherwise; if need arises, these sensors are coupled with Inertial Measurement Unit (IMU) or gyroscopes. Configuration and quality of the components depend on the requirements. ADCS assembly has been sketched in Alg. 3; considered disturbance effects are gravity gradient, solar pressure, aerodynamic drag and torques and main thruster misalignment [GF04]. Two groups of partial solutions are created according to requirements, compatible sensors C_s and candidate actuators C_a; compatible ADCS solutions are assembled from the union of the two subsets. If a single payload has an accuracy requirement more demanding that the others, a solution involving a less complex attitude control and a dedicated stabilised platform is analysed. Following

Algorithm 3 ADCS Assembly

```
Evaluate requirements, estimate disturbances
while C_a do                                   ▷ Cycle on possible actuator types
    Solution()
    if addGNC then Solution += GNC electronic board
    end if
    if addMT then Solution += Magnetorquers
        addMag
    end if
    if addTH then Solution += Thrusters
    end if
    if addCMG then Solution += Control Moment Gyroscopes
    end if
    if addRW then Solution += Reaction Wheels
    end if
    while C_s do                               ▷ Cycle on possible sensor types
        if addSS then Solution += Sun Sensor
        end if
        if addHS then Solution += Horizon Sensor
        end if
        if addMag then Solution += Magnetometer
        end if
        if addST then Solution += Star Tracker
        end if
        if addIMU then Solution += Inertial Measurement Unit
        end if
    end while
    if Solution is Complete then
        Estimate harness
        Add to V
    end if
end while
Search for best solution
return ADCS Solution
```

C&DH multiple-units philosophy, the subsystem has a dedicated electronic board for monitoring and management purposes. Position and orientation for the external elements (sensors, thrusters) is assigned during structures and mechanisms build-up.

3.1.6 Electrical Power System

The EPS provides, stores, distributes and control spacecraft electric power [LW99]. In a conventional-approach satellite the most important sizing requirements are the peak and average demands of the subsystems and the orbital profile. Earth-orbiting satellites are standardised from the available power sources point of view; solar panels for primary generation and secondary chemical batteries for storage. Other approaches (radio-thermal generators, fuel cells, capacitors) currently fail in respond to the years-long life, fail-safe, high power-to-mass ratio and operation flexibility requirements. Alg. 4 uses information about satellite peak and average power during both daylight and eclipse, orbit altitude, eclipse duration and mission lifetime as input for the design process. Fig. 3.4 shows the main subsystems components (green elements are related to fractioned design and will be described within the next chapter). Free variables are:

- Solar cells number and type
- Batteries number and type
- Power distribution unit
- Configuration

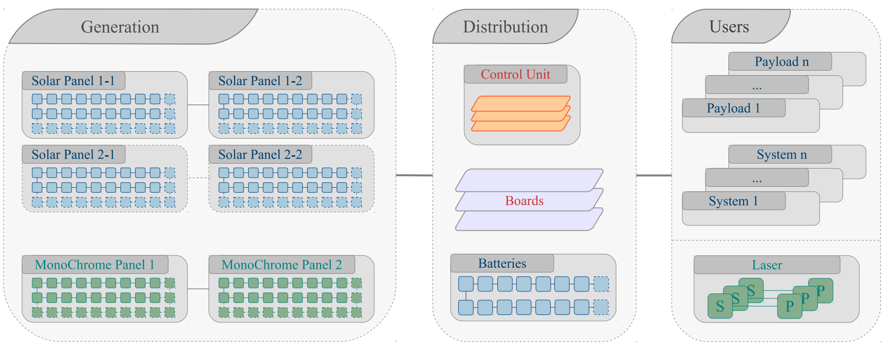


Figure 3.4: EPS main elements

Once again, the process uses database entries for solar cell types (N_s), batteries models (N_b) and power control and distribution modules (N_p). Possible solutions are progressively assembled combining the different components.
P_{sa}, the amount of power that must be produced by the solar arrays, is evaluated according to [LW99, GF04] considering the parameters for cell degradation over the satellite life-cycle, losses of efficiency due to non-optimal work temperature that are typical for every cell type, leading to specific redundancies and initial over-sizing. Similarly battery total capacity, cells number and arrangement (series, parallel) depend both on operative condition and specific features, like depth of discharge or

single cell current. Finally control and distribution boards are selected according to their maximum output and reliability.

Algorithm 4 EPS Assembly

Evaluate requirements
for $i = 1$ to N_s **do** ▷ Cycle on Solar cells
  Solution()
  $n_s = {}^{P_{sa}}/_{power\,solar\,cell_i}$
  if $n_s > 0$ **then** Solution += $(n_s + margin) \cdot solar\,cell_i$
  else Solution = NULL **break**
  end if
  for $j = 1$ to N_b **do** ▷ Cycle on Batteries
    $n_b = {}^{C_b}/_{capacity\,battery_j}$
    if $n_b > 0$ **then** Solution += $(n_b + margin) \cdot battery_j$
    else Solution = NULL **break**
    end if
    for $k = 1$ to N_p **do** ▷ Cycle on Control units
      if $constrain(PCDU_k|power, realibility)$ **then**
        Solution += $PCDU_k$
      else Solution = NULL **break**
      end if
    end for
  end for
  if Solution is *Complete* **then**
    Estimate harness
    Add to **V**
  end if
end for
Search for *best* solution
return EPS Solution

3.1.7 Propulsion

The propulsion system of a LEO satellite ensures the spacecraft capability to perform the initial commissioning from the launcher-release to the operation orbit, maintenance, orbit transfers and, according to in force regulations, end of life de-orbit manoeuvre (or at least a transfer to a safer graveyard orbit). It can also discharges attitude control functions. The subsystem selection and sizing depend on orbital features and the satellite characteristics; ADCS concurrent design influences propulsion design, Alg. 5. Given orbits parameters, required ΔV and thrust levels for manoeuvres and disturbances compensation are calculated [GF04]. Constrains are compared with available thrusters (list includes both mono and Bipropellant Thruster (biTH)) for compatibility. According to comparison results, the under-construction solution can include one (or more) thrusters of different types. Then different tank designs, compatibility with required propellants type and quantity, are investigated (cylindrical/spherical/toroidal shapes, titanium/maraging steel, regulated/blowdown). Finally the solution that, including propellant for attitude thrusters (if any), redundancies and harness, minimises the cost function is selected. Procedure outputs are the

Algorithm 5 Propulsion Assembly

Consider ADCS Solution
Evaluate transfer/station keeping manoeuvres
ΔV budget and thrust level constrains
for $i = 1$ to N_t **do** ▷ Cycle on available thrusters
    Solution()
    $constrainCheck(thruster_i | T_{transfer}, T_{maintenance})$
    if $T_i <= T_{transfer} \& T_i <= T_{maintenance}$ **then**
        $n = {}^{T_{transfer}}/_{T_i}$
        Solution += $thruster_i \cdot n$
    else
        if $T_i <= T_{transfer} \& T_i > T_{maintenance}$ **then**
            $n = {}^{T_{transfer}}/_{T_i}$
            Search for *maintenance thruster*
            Solution += $thruster_i \cdot n$ + *maintenance thruster*
        end if
    end if
    for $j = 1$ to N_p **do** ▷ Cycle on tank material and shape
        Solution$_{i,j}$ = Solution$_i$
        check required propellants $f(thruster_i)$
        Solution$_{i,j}$ += Tank sizing$_j$
    end for
    if Solution is *Complete* **then**
        Estimate harness
        Add to **V**
    end if
end for
Search for *best* solution
return Propulsion Solution

updated part lists and the new contributions to the mass and power budgets.

3.1.8 Thermal Control System

The control of the temperature of the spacecraft equipments and structural elements is required for two reasons [FSS11]: most of electronic and mechanical parts are designed to work efficiently within a narrow temperature range and materials usually have a non-zero thermal expansion coefficient thus meaning that a temperature change introduces a thermal distortion. In order to evaluate with an automated fashion the expected temperature ranges of the satellite several tasks have to be performed:

1. Identification of the heat sources: external (LEO assumed, orbital parameters are known) and internal (from component list, dissipated power of every elements is an input parameter)
2. Creation of a simplified thermal model
3. Identification of the worst hot and cold cases;
4. Selection of passive thermal control components: materials, coatings, paints
5. Evaluation of the components temperature during hot and cold cases

The briefly summarised points involve a considerable effort; point 1 is quite straightforward within the limits of the used simplifications. Point 2 in considerably more complicated; it can be divided into two main operations: construction of the model from a mathematical point of view and the identification of the numerical data required to fill it. The spacecraft has been represented using a lumped mass [IDBL07]; a single thermal node depicts the satellite with its internal heat generation (electronic components losses) and external fluxes (Sun, albedo, deep space). Additional hypothesis have been made to identify the worst design cases:

- Worst Case Hot (WCH), payload at peak power, all other components at nominal power, direct sunlight, albedo and Earth IR
- Worst Case Cold (WCC), all components at standby power and eclipse conditions

The design of the Thermal Control System (TCS), Alg. 6 must ensure that each components temperature is kept within acceptable ranges (minimum and maximum operative with a $15°C$ margin); as the spacecraft is considered as a single body, the tightest combination of minimum and maximum temperature is considered. The aim of the algorithm is to find a combination of satellite coating, radiator and eventually heaters such that constrains are satisfied. The properties of a finite number of coatings N_c (white and black paints, metallised kapton, MLI) are used to evaluate the end-of-life temperatures (WCC and WCH) of the single-node satellite (assuming a uniform coverage). If the constrains are not respected, additional components are added to the solution until design conditions are satisfied (if possible). Purely passive solutions are promoted; however if temperature variation exceed limits, typically due to cold cases, patch heaters can be used to increase dissipated heat, at cost of additional power to be provided by the EPS. The mass increment due to the increased power subsystem complexity is considered during *best* solution selection. Components and renewed budgets are provided as output from the procedure itself.

Algorithm 6 TCS Assembly

Evaluate environmental conditions
Analyse part list for T_{oper}
for $i = 1$ to N_c **do** ▷ Cycle on available coatings
    Solution()
    Evaluate $coating_i$, WCH
    Evaluate $coating_i$, WCC
    if $constrain(WCH, WCC|T_{operation} \pm margin)$ **then** Solution += $coating_i$
    else
        if $(WCH > T_{oper}^{max} - T_{margin}$ **then** design Radiator
        end if
        if $(WCH < T_{oper}^{min} + T_{margin}$ **then** design Heater
        end if
        if $(WCC > T_{oper}^{max} - T_{margin}$ **then** design Radiator
        end if
        if $(WCC < T_{oper}^{min} + T_{margin})$ **then** design Heater
        end if
        if $constrain(WCH, WCC|T_{operation} \pm margin)$ **then**
            Solution += $coating_i$
            Solution += $Radiator||Heater$
        end if
    end if
    if Solution is *Complete* **then**
        Estimate harness
        Add to **V**
    end if
end for
Search for *best* solution
return TCS Solution

3.1.9 Structure & Configuration

The structure has the task to support all other subsystems, providing an interface with the launch vehicle and ensuring the satisfaction of stiffness and strength requirements [FSS11]. Mechanical design has been simplified; a structural idealisation of the satellite has been used. Alg. 7 returns dimension and thickness of the main structure and the configuration of both internal and external components. Required input parameters are:

- Launcher features, minimum axial and lateral frequency, load factors, fairing dimensions
- List of the components with related mass, dimension and type

The structure is assumed to be a 4-sides prism; width and height are evaluated using iteratively an algorithm designed to solve a three-dimensional bin packing problem [MPV00]; given a set of rectangular-shaped boxes the algorithm returns the number of assigned-dimension bins that are required to hold them all. Working in the reverse direction, it has been used to evaluate the minimum dimensions of a structure able to contain all the mentioned elements; inner components are idealised by means of rectangular boxes. A heuristic has been adopted to avoid that the geometry could evolve towards thin beams or, on the opposite, flat plates; side ratio close to fairing geometry are encouraged. Inner elements configuration returns both mass distribution and base dimensions for the structural elements; then mechanical properties of the proposed design are evaluated using a uniform beam model [LW99, GF04]. The

Algorithm 7 Structure/Configuration Assembly

Separate internal/external components
Create internal configuration ($fairing\,dimensions$, $aspect\,ratios$)
for $i = 1$ to N_m **do** ▷ Cycle on available materials
    for $j = 1$ to N_g **do** ▷ Cycle on available geometries
        Solution($material_i, geometry_j, F_{axial}, F_{lateral}, f_{min\,axial}, f_{min\,lateral}$)
        Create external configuration
        Solution += mechanisms
    end for
    if Solution is *Congruent* **then**
        Estimate harness
        Add to **V**
    end if
end for
Search for *best* solution
return Structure/Configuration Solution

monocoque structure is sized for rigidity to meet the natural frequency requirements, applied and equivalent axial loads and tensile strength. Design factor of safety are included; the thickness of the proposed structure must satisfy all the above mentioned requirements. External components position is assigned according to their function in order to avoid interferences. A certain degree of arbitrariness is used during this phase; components are placed along the sides considering their tasks. Using satellite geometric reference system ($+X$ normal to the Earth-pointing side, Z along vertical axis), the standard arrangement is:

- $+X$ Payloads, TT&C antennas and pointing mechanisms
- $\pm Y$ Solar panels (including deployment systems and gimbals), attitude thruster
- $+Z$ Payloads/antennas if needed
- $-X$ Thermal radiator, Star Trackers
- $-Z$ Main thruster(s)

Payloads and antennas placement on $+X$ side has been performed using a 2d version of the bin packing problem solver. The final output of the structure design is a simplified geometry for the satellite frame, a possible interference-free configuration, mass and power required by mechanisms and main structure features.

3.2 Reliability

Mission reliability (the probability that a device will function without failure *that impairs the mission* over a specified amount of time) has been considered during the subsystems design; each of them has been assembled in order guarantee a reliability of .95 at *design life*, the intended operational time on orbit. This has been done by taking into account both components failure rate and their connections (series, parallel, combinations). In order to achieve desired reliability level, both hot and cold redundancies have been used. Trade-offs between high-quality components and less-reliable with backups are evaluated during subsystems assembly or similarly high-performance components opposed to less capable ones. Solutions including the former approach turned out to be favourite during best solution selection for their lower mass; furthermore the latter concept could also fail in finding a feasible solution due to the excessive parallel blocks that have to be added to achieve the required reliability level. Partitioned redundancy [HH85] (elements connected both in series and in parallel) has been used to combine performances and reliability (e.g. 2 series of battery cells with cross-connection to reach both required capacity and ensure that a single element failure does not affect a whole series).

3.3 Costs Model

Cost is an engineering parameter that varies with physical parameters, technology and management methods [LW99]. The cost of a spacecraft system depends, among the other, on its size, complexity, implemented technologies, desired lifecycle, design and political considerations. Cost Estimating Relationship (CER)s are mathematical equations that use regression techniques to establish a relationship between independent variables that are representative of the design, and cost as the dependent variable [boo08]. CERs can be applied at the system level (e.g. spacecraft, instrument), subsystem level (e.g. attitude determination & control, optics) or component level (e.g., star tracker, CCD). All cost models, in their basic form, have some underlying CER defined [boo08]. A variety of tools, both based on CER and with different approaches have been developed, Tab. 3.2; unfortunately none of them has been released for public use [NASc] or they are commercial products [KML05, ADL98]. The formu-

Model	Developer	Spacecraft Estimating	Instrument Estimating
NASA Instrument Cost Model (NICM)	JPL	N/A	✓
Multivariable Instrument Cost Model (MICM)	GSFC	N/A	✓
Space Based Optical Sensor Cost Model (SOSCM)	Aerospace	N/A	Optical Only
NASA/Air Force Cost Model (NAFCOM)	SAIC	✓	✓
PRICE H	PRICE Systems	✓	✓
SEER-H	Galorath	✓	✓
Small Satellite Cost Model (SSCM)	Aerospace	Small Spacecraft	N/A

Table 3.2: Cost Estimation Methodology Examples

lation of a new cost estimation tool was far beyond the main topic of this research; the time required to evaluate and investigate a satellite database in order highlights the necessary CERs would have covered a large part of the PhD program. Furthermore the required databases are created by the same companies or organisations that develop the cost estimation tool, therefore they are not public. Then, the creation of the database would have been the first step. Instead, simplified cost relationships have been retrieved from available literature. The models from [LW99] proved to be antiquated as they include satellite built more than 30 years ago, thus being not so representative of later, improved designs. A more updated alternative has been identified in [KKC12]; the multi-parameters CER engineered by the authors are based on Earth observation satellites and in order to overcome the limitations of mass-based prediction models, evaluation of the system complexity Index and cost correction relationship are also applied to the cost model to increase the accuracy of the model [KKC12]. Always according to the authors, their CERs have better degree of confidence when applied to Earth-dedicated science satellites whose payload features are known. Costs related to launch, initial orbital transfer and commissioning have been calculated using launchers data-sheets. Personnel and ground support costs have been evaluated using NASA's Cost Estimation Toolkit (CET) Software Package [NASa].

3.4 Results

The design tool has been tested using existing satellites with a wide range of payload requirements and orbits. A short resume of their features can be found in Tab. 3.3; the group includes Earth-science satellites built and launched (or scheduled) between 1999 and 2016. Satellites features, both the payload properties used during the design phase

	Total dry mass [kg]	Max. power [W]	Payloads mass [kg]	Years (design)
Aura	1767	4600	1160	2004-2010
CryoSat-2	684	850	90.7	2010-2014
EarthCare	1635	1130	605	2016-2019
IceSat-2	890	850	318.7	2003-2009
Ikonos-2	726	1500	171	1999-2006
Jason-2	525	550	110.7	2008-2013
Proba-1	94	90	17	2001-2003
RapidEye	144	64	43	2008-2015
Sentinel-1	2146	4800	985	2013-2020
Calipso	595	560	185	2006-2009
GeoEye-1	1260	560	452	2008-2015

Table 3.3: Reference satellite main parameters

(SAP spacecraft are highlighted using the *italic* font) and the comparison data have been obtained from [ESAa]; in Fig. 3.5 and Fig. 3.6 design and real satellites dry mass and maximum power are reported. Data has been ordered according to payload mass. Main differences in Fig. 3.5 have been observed for small-size satellites (mainly Proba-1 and RapidEye) and Sentinel-1. Trend curves have been added using exponential interpolation functions. Dry mass graph highlights a similar relationship for mass vs payload mass; SAP spacecrafts on average tend to be 8% lighter. During evaluation, conventional mass and power margins (both for subsystems and satellite) from [LW99] have been used according to the class of the under-design satellite. Maximum power graph shows larger differences, especially for small satellites (SAP over-estimates). Real and designed configurations have been investigated in order to detect the source of the inconsistencies. Analysis highlighted that:

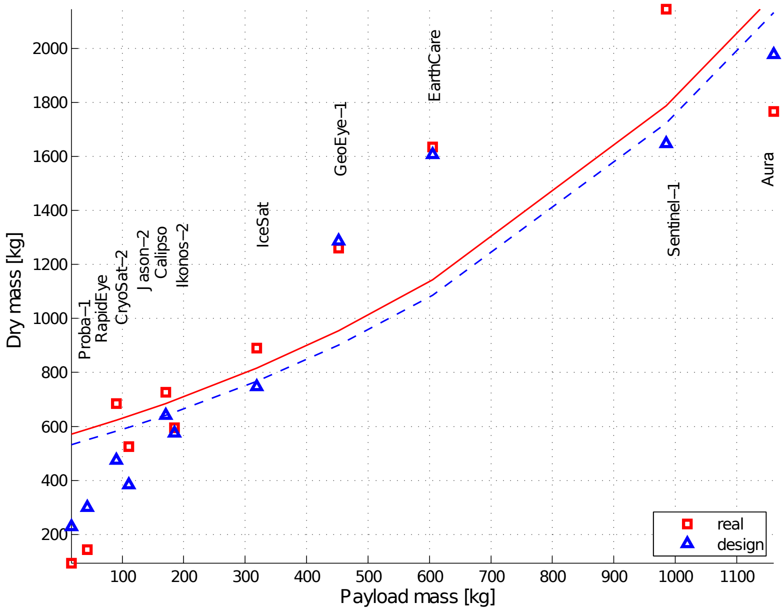


Figure 3.5: Dry mass, designed vs real satellites

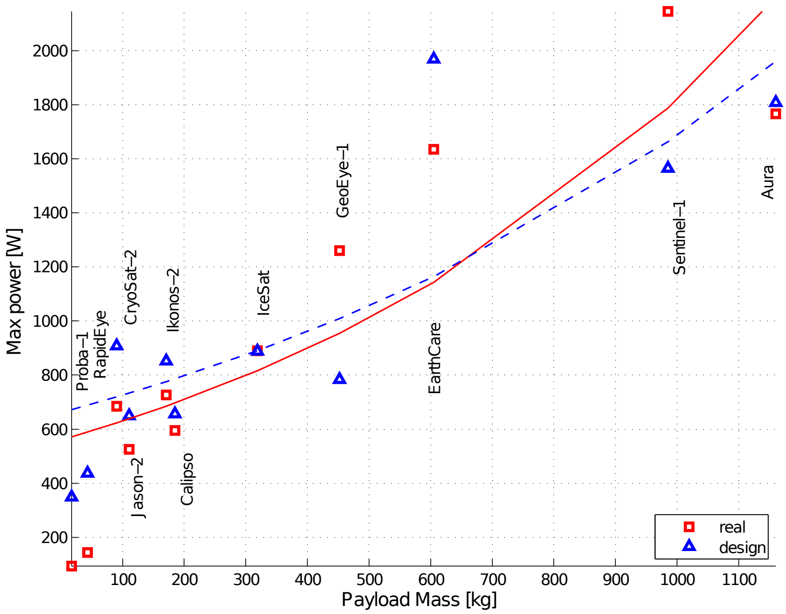


Figure 3.6: Total power, designed vs real satellites

- Proba-1 was a in-orbit technology demonstration satellite, built to test newly designed instruments (as a light-weight, low power ST) and navigation techniques, resulting in a spacecraft lighter and less power demanding spacecraft than its more conventional counterpart *Proba-1*.
- RapidEye is part of a 5 satellites constellation, all commissioned using a single DNEPR launch; the release orbit was similar to the operative one, thus requiring only a Xe resistojet thruster for constellation maintenance. The considered *RapidEye* used a less specialised release orbit, that lead to more complex propulsion system based on a bi-propellant, whereas other subsystems had the same configuration.
- Sentinel-1 difference has been caused by a design life issue: although the official requirement is 7.25 years it carries consumables for up to 12 years, increasing its total mass and making the MT exploit preferable over TH (still carried for dual attitude/orbit maintenance purposes). Extending *Sentinel-1* design life resulted in RW+MT ACS configuration (but also increased the dry mass due to additional redundant components).

General configurations for ADC, TT&C and Propulsion subsystems have been summarised in Tab. 3.4; communication system solutions, in general, are the same, relying on X-band downlink for large volume science data and S-band channels for telemetry/small volume science download and commands uplink. ADS and Propulsion show some diversities; as happened for Sentinel-1 balance between thruster and magnetorquers as desaturation actuators can be staggered by including extended operation time in consumables evaluation. When propulsion requirements are evaluated, the proposed attitude solution that already includes 4 thrusters acting on a single side is considered; if thrust requirements are satisfied the mass of the propellant required (not including extra for life extension) could make the combined ADC/Propulsion system preferable to separated attitude actuator (RW and MT) and dedicated orbital manoeuvres thrusters. Attitude determination for all the satellites (real and re-designed) is based on star trackers and gyroscopes with additional magnetometers for those spacecrafts that have MT actuators; this has been caused by the high accuracy requirements. Similarly foreseen thermal controls are all passive with additional heaters and radiators. As previously mentioned, EPS configuration for Earth-observation satellites is almost standardised with limited variations in solar arrays arrangement. Similarities and differences between the case study satellites and the proposed solutions underlined the current flaws of the proposed procedure:

- passive-restricted thermal control -could be impossible to find a solution compatible with the worst cases hot and cold in satellites with considerable heat dissipation-
- the used structural model based on a uniform beam is unrealistic
- internal configuration is constrained to 4-sides box geometries

	ACS	TT&C	Prop.
Aura	RW+MT	X-band dl, S-band ul/dl	4 hTH
	RW+TH	*X-band dl, S-band ul/dl*	*12 hTH*
CryoSat-2	MT+cgTH	X-band dl, S-band ul/dl	cgTH
	RW+TH	*X-band dl, S-band ul/dl*	*12 hTH*
EarthCare	RW+MT	X-band dl, S-band ul/dl	4 hTH
	RW+TH	*X-band dl, S-band ul/dl*	*12 hTH*
IceSat-2	RW+TH	X-band dl	4x22N, 8x4.5N hTH
	RW+TH	*X-band dl, S-band ul/dl*	*12x10N biTH*
Ikonos-2	RW+MT	X-band dl, S-band ul/dl	?
	RW+MT	*X-band dl, S-band ul/dl*	*4 hTH*
Jason-2	RW+MT	S-band ul/dl	? hTH
	RW+MT	*S-band ul/dl*	*12x5N hTH*
Proba-1	RW+MT	S-band ul/dl	-
	RW+MT	*S-band ul/dl*	-
RapidEye	RW+MT	X-band dl, S-band ul/dl	1 Xe TH
	RW+MT	*S-band ul/dl*	*1 biTH*
Sentinel-1	RW+MT+TH	X-band dl, S-band ul/dl	14 hTH
	RW+TH	*X-band dl, S-band ul/dl*	*12 hTH + 1 biTH*
Calipso	RW+MT	S-band ul/dl	4 hTH
	RW+TH	*S-band ul/dl*	*12 mono + 1 biTH*
GeoEye-1	RW+MT	X-band dl, S-band ul/dl	8x22.2N hTH
	RW+TH	*X-band dl, S-band ul/dl*	*12 hTH + 1 biTH*

Table 3.4: Real vs SAP spacecrafts subsystems

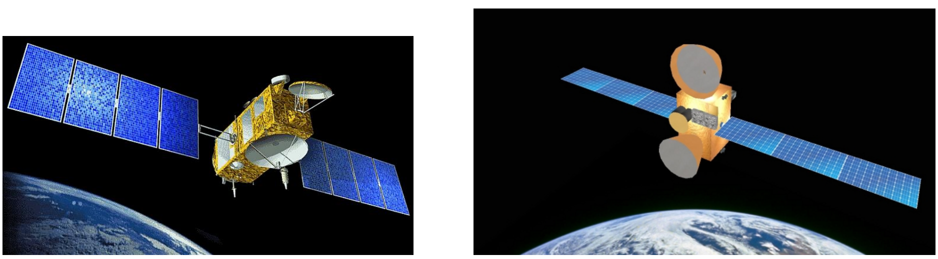


Figure 3.7: Real Jason-2 satellite (left) and SAP version (right)

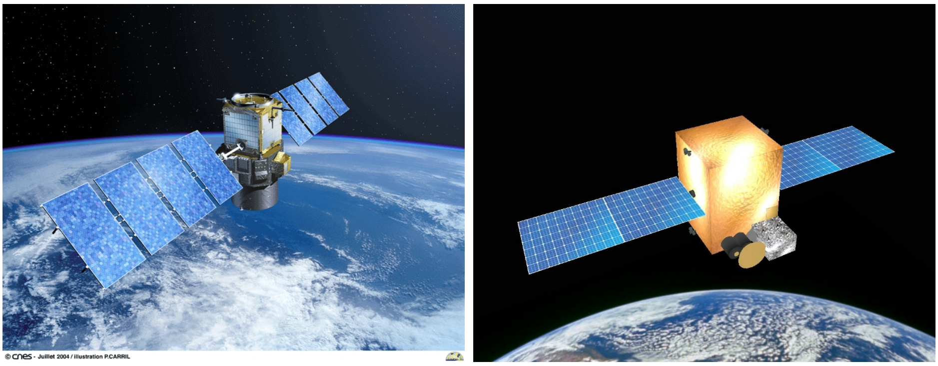


Figure 3.8: Real Calipso satellite (left) and SAP version (right)

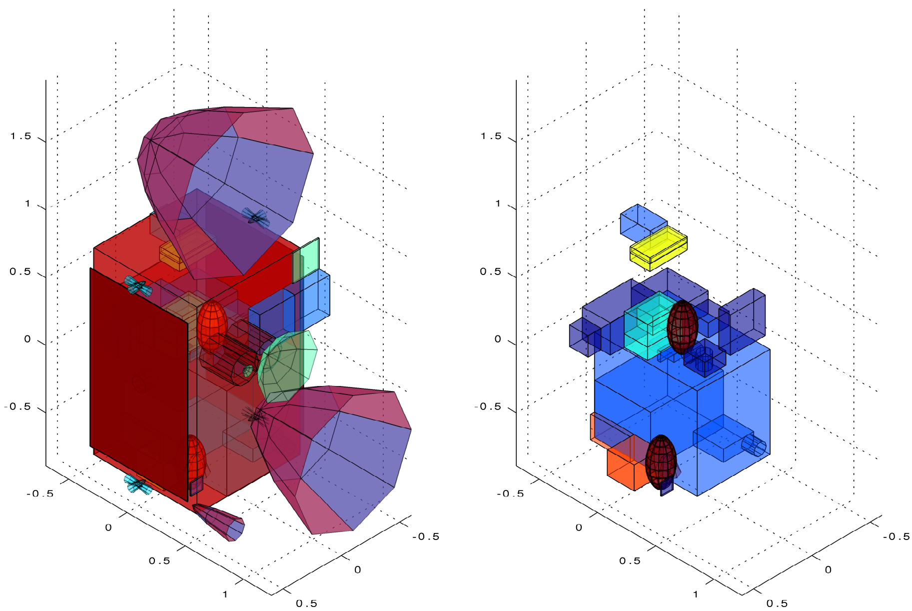


Figure 3.9: Jason-2 SAP version, complete schematic with highlighted internal elements

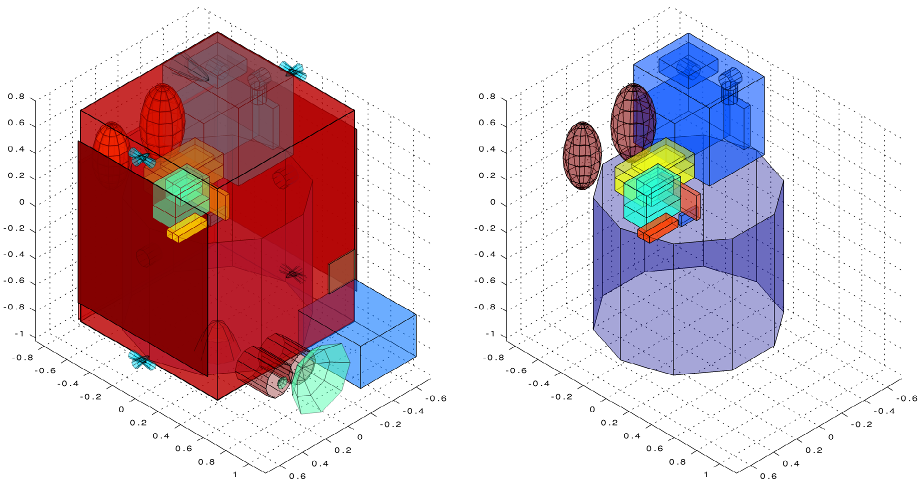


Figure 3.10: Calipso SAP version, complete schematic with highlighted internal elements

- external configuration is driven by heuristics instead of consider all possible topologies
- ADCS design is limited to one class of actuator (two if desaturation is required) with a fixed configuration
- space consumable evaluation could be improved (currently driven only by disturbances and manoeuvres over mission lifetime with no spares)

Solutions achieved for the EPS and TT&C subsystems resulted to be the most accurate in the tool. In spite of the highlighted differences, the test cases show that the SAP module manages the input correctly and proves to be reliable enough to provide preliminary design satellites with an acceptable degree of confidence.

CHAPTER 4

Fractionated Design

4.1 Multiple Spacecraft Assembly Procedure

The Multiple Spacecrafts Assembly Procedure (MSAP) is the highest level of synthesis used in the design procedure of this work, Fig. 4.1. The aim of the algorithm is to engineer a group of satellites able to operate in a collaborative fashion. In order to achieve such result, module-designer must be aware that the under development satellites will be asked to have supplementary hardware and that their requirements are not only dictated by the payload operations but an additional set of constrains must be envisioned. The four main functional blocks that constitute the MSAP are:

- Evaluate Input Parameters (EIP) performs a pre-processing of the mission requirements and handles the fractionation/payloads distribution over different modules.
- SAP given payload requirements and fractionation levels designs a satellite able to satisfy both of them. Mainly discussed in Chap. 3.
- General Constrains Evaluation (GCE) evaluates overall constrains satisfaction.
- System Evaluation (SE) defines performance parameters for the proposed configuration.
- Iteration Loop (IL) part of the optimisation process to search best-performing satellites.

Each functional block has been specifically designed to address one aspect of the optimisation process proving inputs for the single satellite design tool and reading the outcomes in order to adjust the requirements.

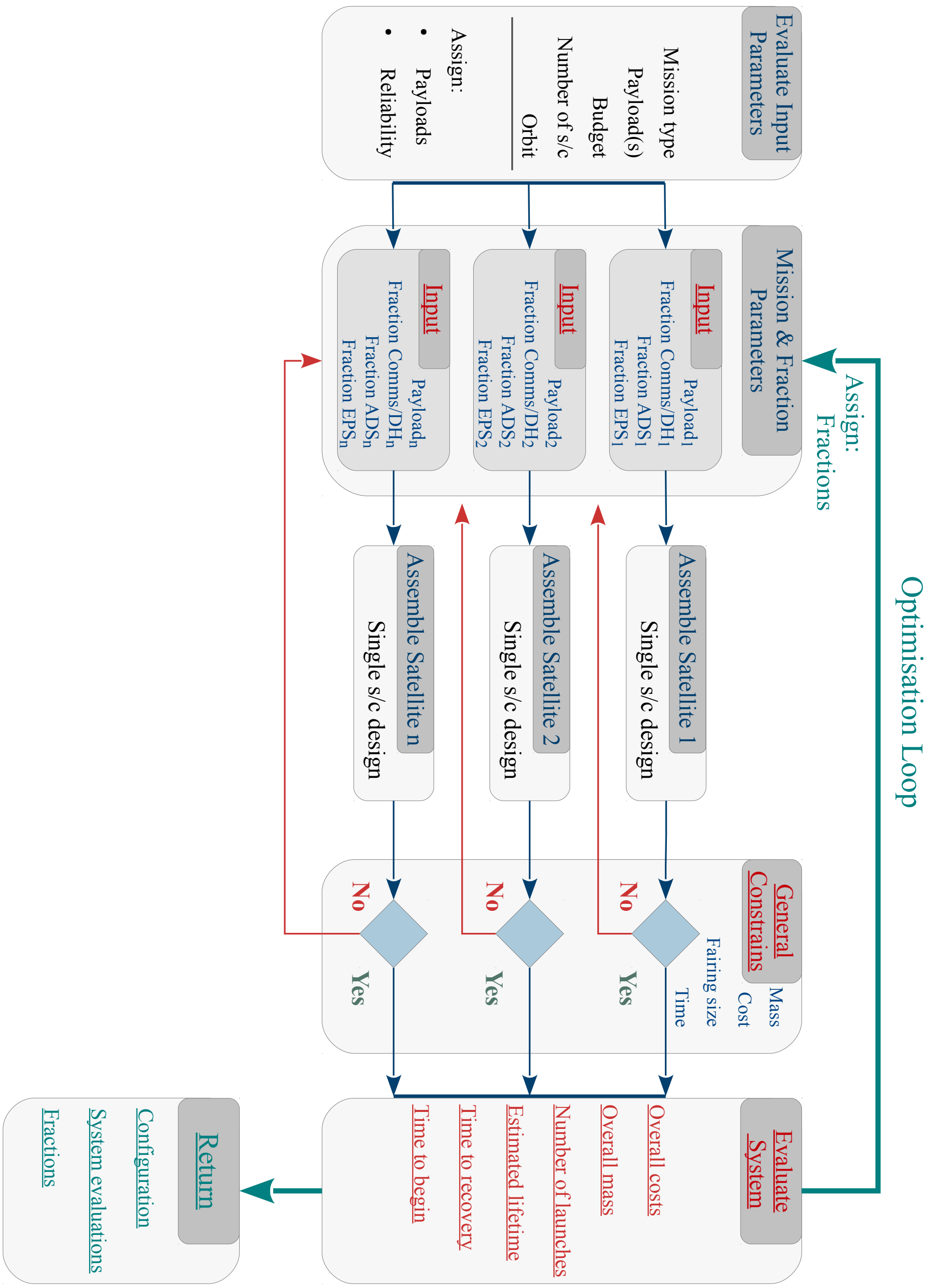


Figure 4.1: Spacecraft assembly procedure

4.1.1 Evaluate Input Parameters

The EIP block has been created to perform two main functions: at the beginning of the design converts high level requirements into a series of input for the satellite assembly procedure. Every data-block contains both constrained variables about payload and orbital features and design parameters describing how the module will contribute to the overall spacecraft by means of its shared resources (master/subordinate positions for every fractionated subsystems, amount of shared resources). Fractionation and number of modules can be assigned or left as free variable, as well as the launch vehicle selection. Subsequently, it changes fraction design parameters according to the feed-backs obtained through the ES and the IL. The input parameters evaluation creates requirements and constrains used to the feed the SAP according to:

- Mission type: main mission objective (as science or communication)
- Payloads: complete list of desired payloads and their requirements
- Fractions: resources that are required to be shared
- Budget: maximum affordable budget
- Launchers: constrains on launchers class, type or number of launches

When multiple payloads and satellites are provided, their allocation is dealt as a combinatorial optimisation problem; payloads are assigned to different spacecraft similarly to the "knapsack problem" [MT90]. The aim is to create input parameters that could lead to similar satellites, possibly increasing common design choices and components; cost saving related to the exploit of a standard platform has been considered. Preliminary mass and power estimations of proposed satellites born from payloads/fraction parameters combinations [Bro03] are used as pay-off metric for a dynamic programming unbounded knapsack problem solver [MT90] (a restricted version of the original problem, based on the hypothesis that all *weight* are non-negative integers). The solution time is pseudo-polynomial on the number of payloads and the result is instruments allocation over different spacecraft. The introduction of fractionation in SAP causes variation in foreseen hardware, operations and consequently on final design result, Sec. 4.2. Multiple instances of the satellite assembly are created, coherently with number of spacecraft to be designed.

4.1.2 General Constrains Evaluation

The GCE holds watch-dog functions over the SAPs; it ensures that the combination of the proposed satellites still satisfies the high level requirements. Used control parameters involve total mass, fairing dimension compatibility and budget issues. A failure in the requirements observance causes the free EIP output parameters (like fractions or number of satellites) that generated the current solution to be changed.

4.1.3 System Evaluation

The SE performs the satellite-cluster-level performance evaluation; its objective is to compare different spacecraft configurations searching for the one that obeys at best to the imposed constrains. Due to the tasks and the structure of the envisioned algorithm, the SE has been integrated with the IL in order to form the base for the optimisation process. Such constrains can be the mass or cost minimisation (or a weighted sum of both); number of launches; minimum construction time spacecraft. A grade is assigned to the achieved solution allowing their comparison and thus providing a numerical fitness.

4.1.4 Iteration Loop

The aim of the IL is to modify input parameters according to the achieved results; is not necessary for a single design whereas takes a fundamental role during optimisation. Due to the non-linearity of the formulated problem [Mos96], identification of cause-effect relationships between the module requirements and the overall spacecraft can be difficult and hardly automatable. A population-based optimisation method has been used to search for the best configuration. The metric and the optimisation algorithm are described in Chap. 7.

4.2 Shared Resources

An essential attribute of fractionated spacecraft is their ability to physically decouple subsystems and payloads by placing them on different modules and, in doing so, enable the sharing of subsystem resources amongst modules via collaboration [BE06a]. Dispersion and subsequent sharing of certain subsystem resources and functions requires additional hardware both on the modules that provide the resources (Source Satellite (SS) or *master*) as well as those modules that rely on/receive the resources (Recipient Satellite (RS) or slave). The hardware associated with each shared resource may be simple instantiations of current technology, as is in the case of distributed communication systems, or could require the application and demonstration of new(er) technologies, as is the case of moving electrical power from one satellite to another. Fractionation could, at least theoretically, be applied to almost every subsystems (structures for obvious reasons are not prone to be shared); various existing, under development or under study technologies could directly or indirectly be applied to this mean. A priory analysis of the available or under development technologies that could be used to connect remote satellite subsystems have been performed in order to select those whose features were compatible with the undergoing study.
A remote TCS would try and regulate the temperature of another spacecraft; temperature evaluation functionality could also be envisioned but would be limited to surface analysis, thus providing limited information about inner elements status. Thermal

control could be achieved by providing energy to the *dependant* spacecraft in the form of concentrated sunlight [SL89], laser [Gla68], or microwaves [Die80]; the main advantage would be a reduction in slave satellite power request (removing or reducing heaters requirements) counterbalanced by a considerably larger increase in SS mass and power due to the reduced efficiency of the energy transfer process. The cost-benefit of a cooling system such as using one spacecraft as shield to protect the subordinate from the direct sunlight would be extremely disadvantageous due to the complexity of involved orbit and attitude control.
Remote propulsion aim is to generate a net force on an object without need for energy source or reaction mass (possibly both) on the object itself. Several methods have been proposed, ranging from beam-powered approaches to magnetic repulsion. Beam-powered propulsion can be further classified into laser [MF06], microwave [Par06], concentrated sun-rays [NKM+05]. In either cases, the main assumption is that momentum may be transferred to a spacecraft by promoting mass expulsion as in a conventional rocket. Final spacecraft velocity is still limited by the rocket equation, but the objective is to reduce non-propellant mass and achieve high specific impulses. Beams can be focused to specific component (absorber cavity or heat exchanger) where energy is transferred to low molecular weight propellant; ablative propulsion in which an external pulsed beam is used to burn off a plasma plume from a solid metal propellant has also been proposed [PBL+10]. Magnetic repulsion can be used to maintain a given distance between two satellites; a conventional propulsion system is still required at least on one of them. Contact-less interaction similar to magnetic suspension is used to move the desired spacecraft. Involved problems are the complexity of the required hardware, limited control capacity, additional requirements for tight attitude and position control. At the state of the art, remote propulsion has a low Technology Readiness Level (TRL), reaching at best 4 with a ground prototype of solar thermal rocket. In force of this consideration, remote propulsion has not been considered in this work as his exploitation word require hardly accurate performances extrapolation and mass or power estimations. A considered alternative is a combination of on-board electric propulsion system alimented through a wireless power source. Pulsed plasma thruster, high-specific-impulse, low-power electric thrusters [Bur98] have been selected as possible propulsion elements and integrated within the SAP.
Attitude control is achieved by providing the spacecraft the capacity of actively or passively modify its own angular momentum; traditionally this is obtained using actuators or acting on satellite mass distribution. Remote attitude control can be performed using a satellite to change and regulate angular velocity and orientation of another spacecraft; interactions can be with or without contact. In the former case a docking/berthing manoeuvre is required, thus the attitude of the two spacecraft is changed and finally the non-controlled satellite is released; supplementary hardware for docking, relative attitude sensor, advanced GNC algorithms are required, making it a technically possible (the privately-founded Mission Extension Vehicle is

based on a similar concept) although not convenient for continuous manoeuvres. The latter concept, contact-less attitude control could be performed constantly altering the strength of a magnetic field produced by electromagnets using a feedback loop. Permanent magnets, with electromagnets only used to stabilise the effect could be used to reduce the power requirements [Kon02]. Similarly, electrostatic charges can be used; current TRL is paused at ground experiments. Contrariwise, remote attitude determination has been tested and qualified, Sec. 4.4. The four remaining subsystems, TT&C and Data Handling (DH), Attitude Determination and System (ADS) and EPS have already demonstrated space-qualified remote operation capacity (as for communication and data handling) or like wireless power have a well established and documented technical background and despite the lack of space-environment tests proved its potential with field experiments (TRL 6). A more detailed description of the selected fractionated subsystem is provided in the following sections.

4.3 EPS

The power subsystem consists of two main elements: power generation and storage (distribution, control and regulation are assumed implicit). These elements could be shared if an effective way to transmit power without physical connections (i.e. cables) is provided. In the case of shared power generation, a *master* satellite in a fractionated spacecraft satisfies its own power requirements plus a part (or almost all) of the power demand of one or more other modules. As a result, these *dependent* satellites have reduced power generation requirements, as they have to produce less (or none of the) power than they require. In a similar fashion, shared power storage consists in design a satellite with a storage capacity able to sustain both itself and the target element that will be free to use smaller storage devices. Power is a fundamental element in satellite design; energy production and storage capability is the base for all electronic components operation. Although fractionated approach foresees that all required power generation and storage could be distributed, this would lead to a non-fail-safe design: whenever the main satellite in the formation would had a malfunction, not only science objectives of slaves spacecraft would be jeopardised but subsystems operation would be endangered by the lack of power. To avoid this criticality, an additional requirement has been add to designed spacecraft: every spacecraft has to be able to provide by itself enough power (and consequently storage) in order to execute basic operations like attitude control, power regulation, data handling, telemetry download. With this distinction, shared power is used to feed only navigation and science related operations.

Even if the conceptual design of the power share is pretty straightforward, its implementation encounters several technical problems, mainly related to the low efficiency of the whole process. According to the distance between the emitter and receiver satellites, two main classes of contact-less power transfer strategies arise:

- near field (a meter or less) $\longrightarrow$ electrodynamic induction, electrostatic induction, resonant inductive coupling
- far field (up to kilometres) $\longrightarrow$ microwave, laser

All the above mentioned techniques rely on electromagnetic waves, the distinction is due to the effective separation in relation to the used wavelength (near $\leq 1\lambda$, far field $> 1\lambda$). Near field techniques have shown during experiments that the potential efficiency can be high (up to 95% and higher in ideal cases), however they require accurate positioning of transmitter and receiver elements; when distance and orientation deviate from nominal conditions, efficiency rapidly decreases [SLR+09, BS11]. Focusing the attention on the far field methods, both of the two technologies are currently investigated for their potential aerospace applications [NK10]. They share the same philosophy, a power generation unit converts electric energy into a focused, high power-density, directional beam toward a target device able to convert this energy back to its electric form [NFR+11a]. In the microwave transfer the generator unit is a microwave emitter (like a magnetron or a klystron) and the receiver can be a rectenna or a hemispherical antenna [RyLC04]; with the laser approach a solid state laser is used to illuminate a special (optimised to work on a single wavelength) photo-voltaic panel [BKB10]. The methods also share the same drawbacks, mainly beam pointing problems, diffusion over long ranges and low overall efficiency (literature provides several estimations, on field 20% [NK10] DC-DC efficiency can be expected). For LEO applications, satellites rely on solar arrays as main power source; low efficiency energy transmission results in large areas for the transmitter spacecraft. In order to concentrate, aim and steer the beam, focusing and pointing devices are required, thus introducing additional constrains on attitude and configuration. Additional thermal problems arise both due to increased solar array surface as well as due to dissipated power from emissive devices. Main features of laser and microwave beam technologies have been reported in Tab. 4.1. Microwave beam has additional losses due to RF

	Laser	Microwave
DC-RF converter	Solid state laser 50%	Magnetron/Klystron 83%
Pointing	Mirror 99%	Phased array antenna 90%
RF-DC converter	AlGaAs photovoltaic cell 59%	GaAs diode rectenna array 82.5%

Table 4.1: Laser and microwave power beaming comparison

filter insertion, beam coupling, propagation, collection and rectenna efficiency, DC to utility grid efficiency that reduce the DC-DC efficiency to less than 45% [Dic03]. Similarly, laser efficiency is limited by beam diffusion, pointing inaccuracies, receiver panel inclination and collection efficiency; a global η of 25% has been estimated [NFR+11b]. According to an a priori efficiency estimate, microwaves should be selected as remote power transfer technology. However, one of the aim of the fractionated design is to remove non-payload related requirements (or at least reduce their influence) from satellite design; rectenna arrays size and relative orientation constrains are more demanding than the laser-photovoltaic cell counterpart, especially if, to increase receiver efficiency, a hemispherical reflector is used. Furthermore, laser beaming offers a series of design and operation advantages:

- Collimated monochromatic wave-front propagation allows narrow beam cross-section area for transmission over large distances
- No radio-frequency interference to existing radio communication (nonetheless, optical instruments must be adequately screened)
- Current technical maturity and undergoing improvements in solid state lasers efficiency (prototypes have achieved wall η greater than 80% [PREZ07])

Based on these factors, laser power beaming has been selected to be employed in fractionated spacecraft that share the power resource. Considering possible (and desired) increases in components efficiency, fractioned satellites have been designed considering variable levels of DC-DC efficiency, ranging from 25% (current technology) to 40% (achievable with under development laser and Vertical Multi-junction photovoltaic cells [NFR+11b, PREZ07])

4.3.1 Satellite Assembly Procedure variations

When fractionation is introduced, power production and management requirements can be increased/decreased according to designed role, Tab. 4.2; *master* modules have to face the challenge of transfer power to the target satellite. This leads to increased solar panels surface and battery capacity, heavier and complex power distribution, laser and control unit (with associated thermal control). As in [BKB10] due to limited available data about high power laser for space application, Commercial Off The Shelf (COTS) continuous wave solid state cutting lasers with liquid cooling systems have been used as baseline; pointing has been provided using specification of communication optical heads and electronic units (a comparison with laser-based science instruments resulted in similar accuracy, mass and power requirements). The *subordinate* on the other side has reduced nominal requirements leading to batteries and body mounted solar panels to satisfy minimum survival needs coupled with panels supporting monochrome cells to convert the laser beam. At SAP input level, shared EPS has been controlled by the continuous parameter $fract_{EPS}$ whose range is ± 1; 1 has been used to identify a configuration where the firstly designed satellite (that assumes the role of *main*) acts as *master* for the rest of the formation, whereas -1 represents the opposite condition, the *main* is served by the other spacecraft. The same parameter also controls the amount of power that will be interested by the remote operations:

	Master	Subordinate
Generation	all internal+ $\sum_{N_s} nav, science$	fundamental subsystems
Storage	eclipse, internal+ $\sum_{N_s} nav, science$	eclipse, fundamental subsystems
Tx/Rx power	$\sum_{N_s} nav, science$	navigation, science
Hardware	Laser generator(s) + Laser head(s)	photovoltaic cells support panel(s) and mechs.

Table 4.2: Shared EPS requirements and hardware

the range ±1 has been mapped on the interval $\sum_{N_s}\{nav, science\} \div science_{main}$. According to this distribution, positive values correspond to designing the *main* with shared EPS hardware able to deliver all (1) or a fraction (< 1) of the power used by the scientific instruments installed on the other satellites; likewise input parameters for those modules will be modified in order to reduce the portion of payload operations that they have to support. And contrary-wise values smaller that 0 are associated to the inversion of the roles, the *main* has become the subordinate element that receives from the other(s) spacecraft; 0 itself identifies a non-fractionated EPS.

4.3.2 Effects on mass and power

EPS fractionation has a massive influence on overall system mass; Fig. 4.2 shows the evolution of system total mass (sum of all satellites) as a function of payload mass for different configurations and dc-dc efficiencies. The reference line (red) represents the monolithic configuration; lines have been interpolated using the same satellites used for SAP validation. Curves for number of satellites greater than 1 have a base offset due to system duplication plus an increment due to the enlarged power requirements; as expected the lower the efficiency, the higher the total mass (and the slope of the curve). Furthermore low efficiency also result in unfeasible solutions (there is no curve for N = 4 and $\eta = 25\%$); encountered design problems arise from:

- EPS, total power can overcome available distribution and control units and solar array drive assemblies capacity
- Thermal, dissipation problems due to both laser and large solar panels area
- Attitude control, actuator sizing
- Structure, configuration and fairing fitting issues

Normalised graph highlights that system duplication is substantial for small satellites, with a mass increment between 160% and 190% with respect to the conventional spacecraft; medium to large satellites under fractionation concept usually result in a single large spacecraft coupled with small modules, thus limiting the overall mass

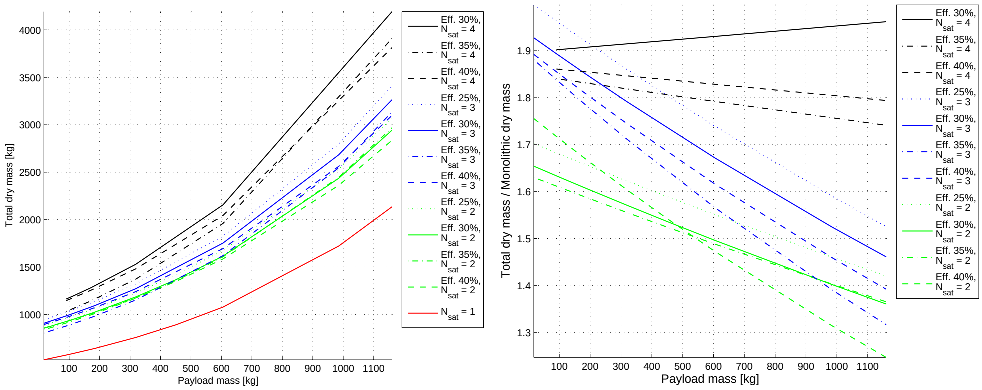


Figure 4.2: Satellites total mass as a function of nominal payload mass and EPS configuration

increment. Fig. 4.3 displays the increased total power requirements under different configurations. The increment is extremely significant, even assuming hypothesised efficiency of 40% the requirements are 3 times as demanding as the original satellite. Fig. 4.4 and Fig. 4.5 report the amount of power generated by solar arrays and bat-

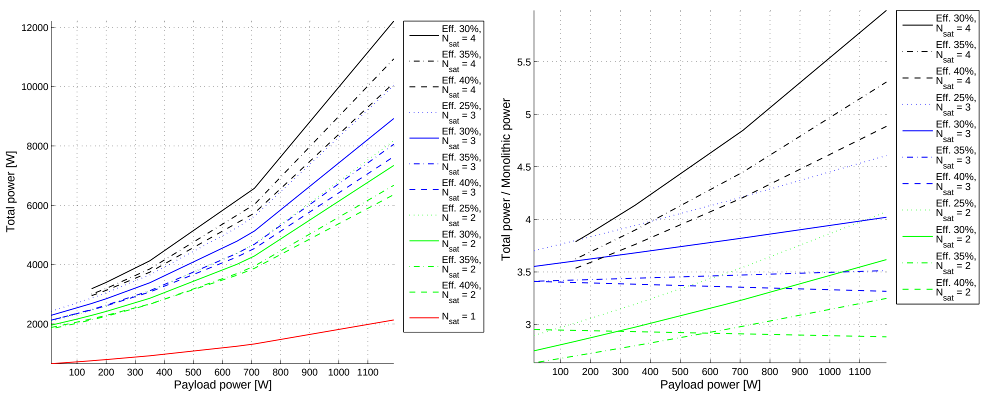


Figure 4.3: Satellites total power as a function of nominal payload power and EPS configuration

teries capacity; Normalised graphs show that due to shared power, the main satellites requirements are extremely more demanding than the original ones. The variation of the estimated EPS subsystem mass and power is depicted in Fig. 4.6; x-axis report nominal satellites mass and power. The comparison highlights that when beam power coupling is introduced, the original power design requirements are shadowed by power transfer needs.

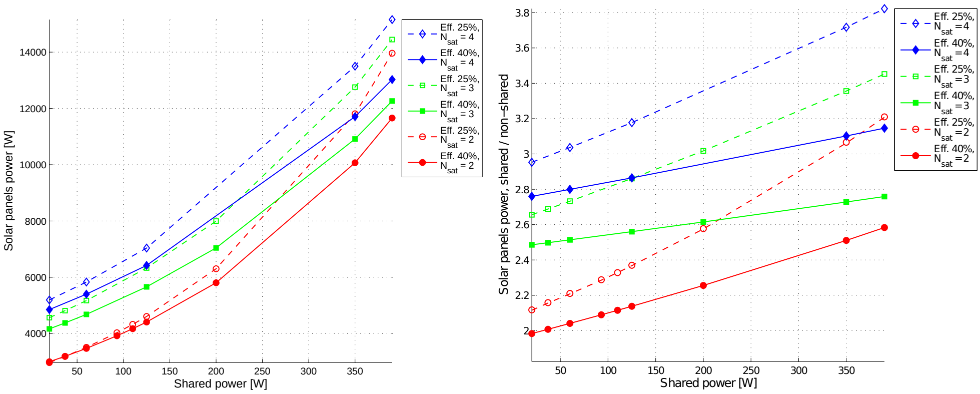


Figure 4.4: Satellites total solar arrays power as a function of shared power and EPS configuration

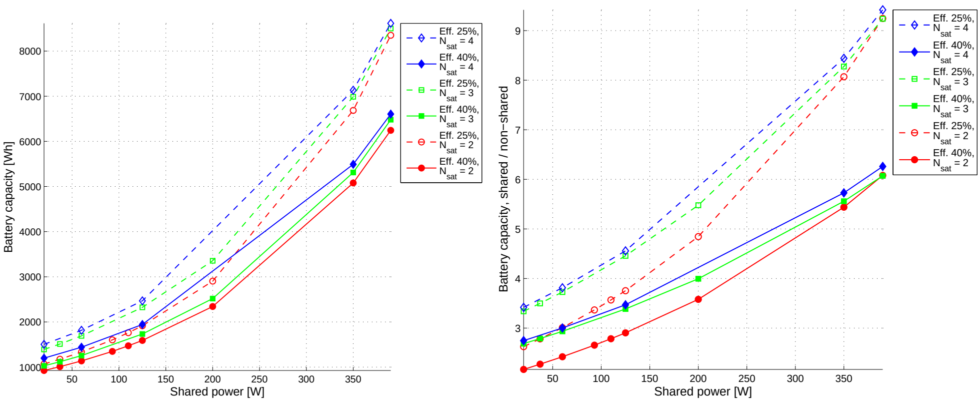


Figure 4.5: Battery capacity as a function of shared power and EPS configuration

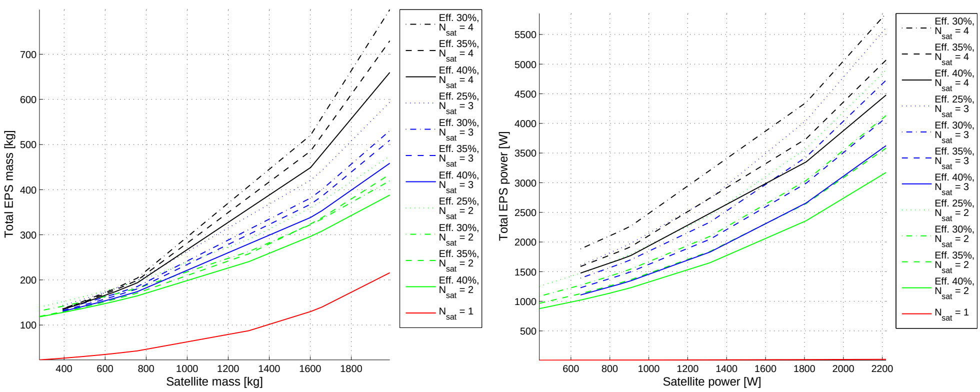


Figure 4.6: Satellites EPS systems as a function of shared power and EPS configuration

4.4 ADS

Attitude determination can be thought to be shared between a couple (or multiple) of satellites; the first spacecraft should be able to evaluate both its attitude (according to an absolute reference) and the relative orientation of the second satellite with respect to its own (movable) reference system. With this knowledge is possible to calculate the attitude of the second element in the global reference; by exploiting a similar concept, if the second module can evaluate the relative attitude of the first module, it only needs the global orientation information of the first spacecraft in order to be able to identify its own absolute attitude. The shared ADS relies then relies on two features: relative attitude determination and inter-satellite communication. So far ADCS fractionation has been restricted to the sole attitude determination; in spite of the existence of proposed remote attitude control technologies (mainly based on electrostatic or electromagnetic fields [Ahs07]) due to their low TRL (˜2) they have not been considered so far (within literature highly variable -if not conflicting- technical specifications and performances have been found). Sensors selection is also affected by the position help by module itself: as the *master* satellite is responsible both for determining its own absolute state vector and evaluating the position of the functionally connected modules, it carries a traditional instruments complement coupled with sensors dedicated to the evaluation of the relative attitude/position. The first function is achieved by different combinations of STs, IMUs, Sun and magnetic-field sensors (according to pointing accuracy requirements); the latter operation can be performed using various techniques; optical (CCD) [HM93], LIDAR [PG04], radiofrequency based [MG13] hardware and software have been already successfully used. High resolution cameras have been selected as possible approach for relative state determination; advantages of this method include low power consumption, no need for dedicated hardware on chaser satellite (small markers could be introduced with minimum mass increase), adaptability to different targets [MSOM07, MBSA06]. *Subordinate* ADCS modules are not strictly required to be able to evaluate their atti-

	Master	Subordinate
Required absolute state accuracy	Highest of all PLs	-
Required relative state accuracy	Highest of remote PLs	Highest on-board PLs
Hardware	IMU+ST multiple VPS	IMU VPS(s)

Table 4.3: Shared ADS requirements and hardware

tude autonomously; this implies that the (expensive) star tracker can be avoided and could be replaced with a simpler virtual positioning system (to have them in both satellites gives a redundancy in case of failure of one of the two). Inertial measurement units are foreseen not only on the *master* element, but also on the *dependant* module, in order to:

- provide a more continuous and responsive attitude propagation capability
- allow attitude determination when the remote system is temporary unable to operate; this could be caused by periodic eclipse phases.

Furthermore two more criticalities are introduced by the remote attitude determination concept, both related to its non-fail-safe architecture: a failure in the main satellite would result is a complete loss of the attitude evaluation for the whole system, unless a backup, possibly a coarse sensors complement as magnetometers or Sun sensors is installed on each slave satellite. This solution would marginally increase system complexity and cost but would allow for a limited operational capacity even in case of subsystem deficiency on the main satellite. The other open point is raised from the necessity to communicate the absolute and relative orientation of the satellites among them: a short range link is required in order to accomplish this task, so similarly to remote data handling, the remote attitude determination introduces a constrain on the communication subsystem. The designed link should ensure close to real time data transfer, with the widest spatial coverage in order to avoid dead zones and have embedded redundancy. Such subsystem could have a non negligible impact on the overall TT&C architecture as well as on the satellite configuration. In the subsequent analysis, the connection between remote attitude and communication has not been investigated properly.

4.4.1 Satellite Assembly Procedure variations and effects

The modified design parameters affect the SAP by changing the accuracy determination requirements and thus the installed hardware. Nominally, the ST are the most expensive and massive attitude determination instruments [Bir96, J.T86], followed by inertial platforms, so replace them with simpler systems would be interesting. The modified SS has a traditional ADS with up to 6 (one per each side) groups of cameras; clusters of optical heads instead of a single one are justified by the needs of both resolution and width of the field of view. Camera model has been retrieved from [O'N10]. Cameras placement poses a problem of its own: the line of sight should be clear from obstacles like other instruments, antennas, solar panels; optics need shielding from direct or refracted sunlight as well as rocket plumes. These are the same requirements that must be respected for ST and payloads, exacerbated by the number of cameras (opposed to the single-dual star tracker heads). As reported in Tab. 4.3, fractionated design applied to attitude determination introduces differences in *master* and subordinate satellites; the effects on mass and power are reported in Fig. 4.7 and Fig. 4.8. Unlike what happened for shared EPS, given a fraction configuration, the mass increment is almost constant regardless the payload mass (that can be broadly considered as an indicator for the satellite complexity); this is due to the fact that cameras arrangement is the same for all spacecraft, so the mayor influence in power and mass graphs is caused by subsystems duplication. Figures depict the two possible approaches to remote ADS: Visual Positioning System (VPS) only on the *master* satellite with a short range communication link with the *served* to inform it about its attitude and VPS on both spacecraft. The former solution turned out to be mass and power saving; the latter allowed a higher redundancy and operation flexibility and could work with a reduced complements on camera sensors (less than one group of cameras per side per module). Normalised graphs highlight how fraction ADS using cameras introduce large penalties on small satellites. Fig. 4.9 reports how different fractionated geometry affect the total to payload mass ratio; once again the smaller the satellite, the larger the influence of the service part of the satellite on the overall system, especially when the spacecraft is composed of multiple modules. As

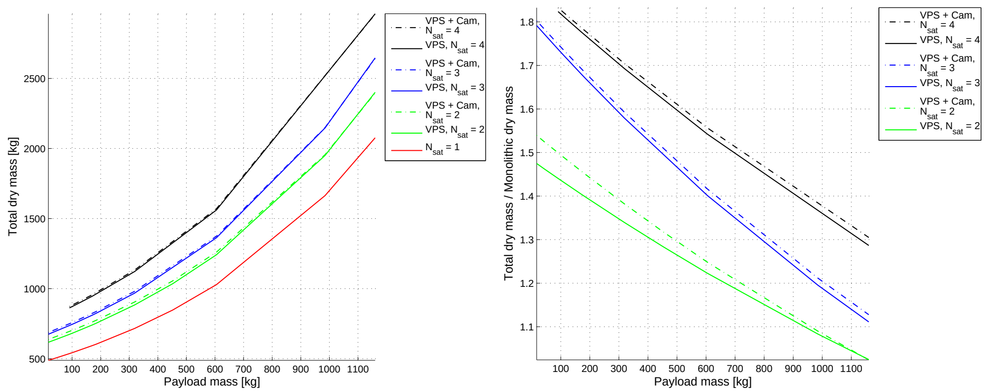


Figure 4.7: Satellites total mass as a function of nominal payload mass and ADCS configuration

anticipated, these analysis have been conducted disregarding the influence of the ADS on the communication subsystem; even so, the remote ADS does not allow neither for a performance improvement nor for a mass reduction and the introduction of the inter-satellite connections are very likely to further increase system's complexity (and so total mass and cost).

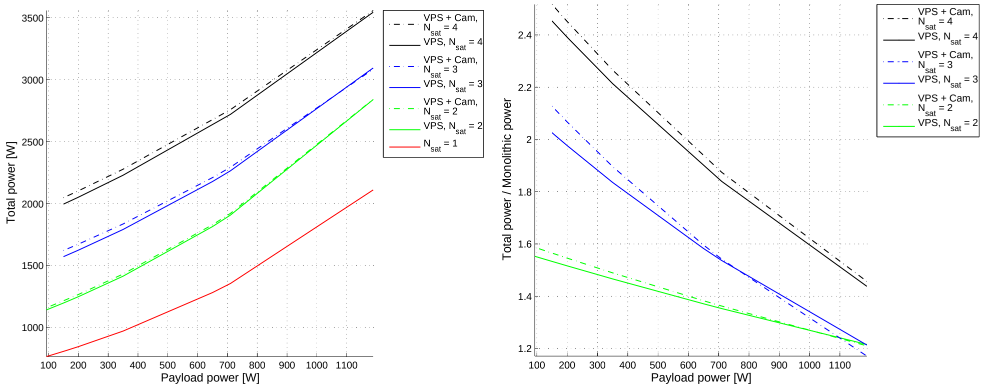


Figure 4.8: Satellites total power as a function of nominal payload power and EPS configuration

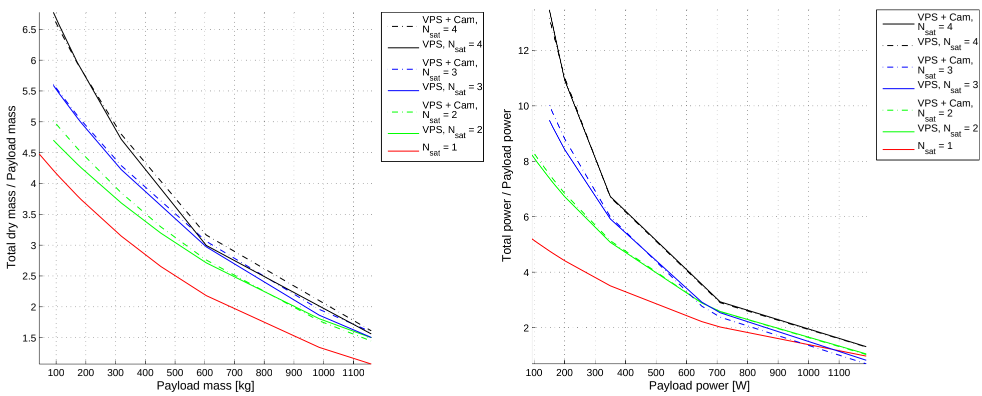


Figure 4.9: Satellite total mass and power increments from monolithic spacecraft according to different ADCS configurations

4.5 TT&C-OBDH

Remote TT&C and DH are here intended as the spacecraft capacity to send (or receive) data to (or from) another satellite, thus enabling memory and computational capacity sharing [BE06b]; the two subsystems have a close relationship as shared storage can be performed only if a way to transmit data across satellites is provided, and likewise the capacity to communicate between satellites is useless if the receiver module has not enough storage to record the information. This is the most intuitive form of wireless connection and in the shape of geostationary communication satellites used as relay for other spacecraft, like the TDRS and the European Data Relay System (EDRS), it has been tested and used for the last 20 years. As with the conventional approach, at least one of the vehicles must have a dedicated system for the uplink and downlink of mission and housekeeping data to a ground station or similarly to an external data-relay satellite -like the just mentioned TDRS-. However, when multiple satellites are available, it may not be necessary to have this kind of antenna on every module, as this creates unwanted redundancy. Omni-directional antennas can be used for inter-module communications to transfer data to and from all modules in the fractionated satellite [O'N10]. In sharing the subsystem resource according to this strategy, a reduction in the requirements induced by the TT&C for the modules without a dedicated ground-downlink capable antenna is achieved as the short-range links demands less power and smaller, cheaper components. Tab.

		Master	Subordinate
Downlink	To Ground	$\sum_{N_s} science, telemetry$	$telemetry_i$
	To Sat	$\sum_{N_s} commands$	$science_i, telemetry_i$
Uplink	From Ground	$\sum_{N_s} commands$	$commands_i$
	From Sat	$\sum_{N_s} science, telemetry$	$commands_i$

Table 4.4: Shared TT&C requirements

4.4 briefly depicts how TT&C requirements have been modified by fractionation: the *Master* satellite covers the task of communication hub, being the only one able to download large volumes of data to the ground station(s). The downlinked data are the sum of internal payloads and telemetry as well as the output of the payloads on the other modules. In order to guarantee a fail-safe condition, even *subordinate* elements have a (limited) capacity to establish direct ground connections in order to receive commands as well as to download their own telemetry in case of unavailability of the main spacecraft, although the exploitation of the comms hub is the preferred way to connect the modules with the command centre. In a spacecraft with non-uniform distribution of computing capabilities amongst the modules, data can thought to be moved among modules to optimise used memory or to perform computationally expensive operation in dedicated modules. A possible application of this concept is the exploit of a high performance computer in a module different from the one that holds the payload, to pre-process raw science data reducing their volume before they are send to Earth. As for the communications, autonomous operations of every satellites must be ensured even in case of critical failure of the main element: housekeeping functions require a dedicated hardware even if the modules holds the *subordinate* position, Tab. 4.5. Fractionated data analysis is then limited to payload related aspects. The percentage of shared memory and processing capacity can be determined during

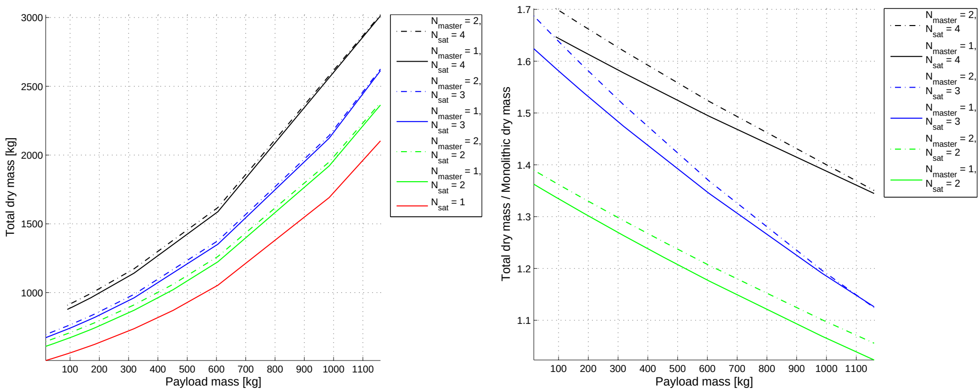
	Master	Subordinate
Memory	$housekeeping_i +$ $\sum_{N_s} science$	$housekeeping_i$
Processing Cap.	$housekeeping_i$ $\sum_{N_s} science$	$housekeeping_i$

Table 4.5: Shared DH requirements

design process: this resources can be easily divided in order to equally share them among different modules or on the contrary concentrate them on a dedicated satellite. Sharing the highlighted resources mostly relies on existing and tested technologies and procedures. The associated hardware for both the *master* and *subordinate* modules is relatively mature; nonetheless there are still notable open points to be addressed, particularly techniques, methods, algorithms, and protocols to ensure the successful management of data delivery, command and data handling, housekeeping and mission data processing, and tasking, scheduling, and control.

4.5.1 Effects on mass and power

Differently from EPS fractionation, DH & TT&C fractions introduce a mass and power increment that is only given by satellite duplication: as it is possible to see in Fig. 4.10-a the curves for different number of satellites exhibit the same behaviour. The main distinction is the offset, caused by subsystems duplication within the spacecraft system. Fig. 4.10-b also highlights that dual hubs configurations (a cluster where two satellites have direct Ground-communication capacities) are penalised when compared to their simpler single hub counterparts. However in the normalised graph, for larger spacecraft, the two options lead to similar results; this is caused by the lower influence on the TT&C on the total mass. Power-based graphs, Fig 4.11, basically confirm the same trend. Fig. 4.13 illustrates the mass of the requited communication subsystem for different test satellites in different fractionated configurations; once again dual-hubs configurations have an higher total mass. The effect is produced by the higher complexity of the subsystem that counts two high gain antennas and communication links (whereas a local connection would require less power and would be lighter). The same data are also reported in Fig. 4.15 using bars, grouped ac-



(a) (b)

Figure 4.10: Satellites total mass as a function of nominal payload mass and TT&C configuration

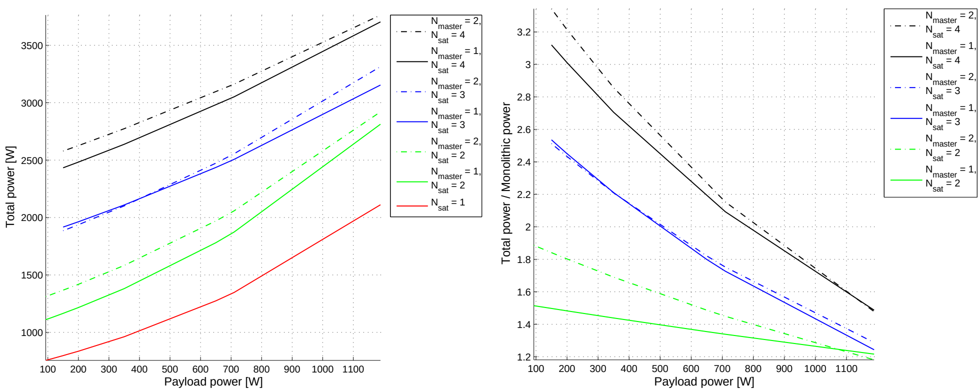


Figure 4.11: Satellites total power as a function of nominal payload power and TT&C configuration

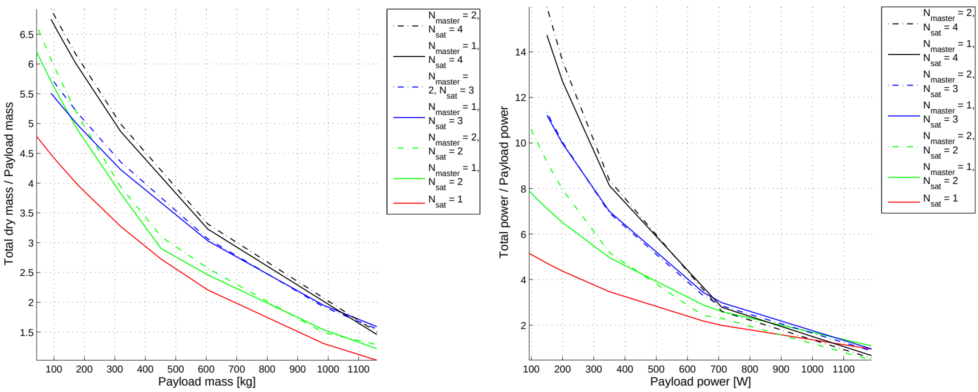


Figure 4.12: Satellite total mass and power increments from monolithic spacecraft according to different TT&C configurations

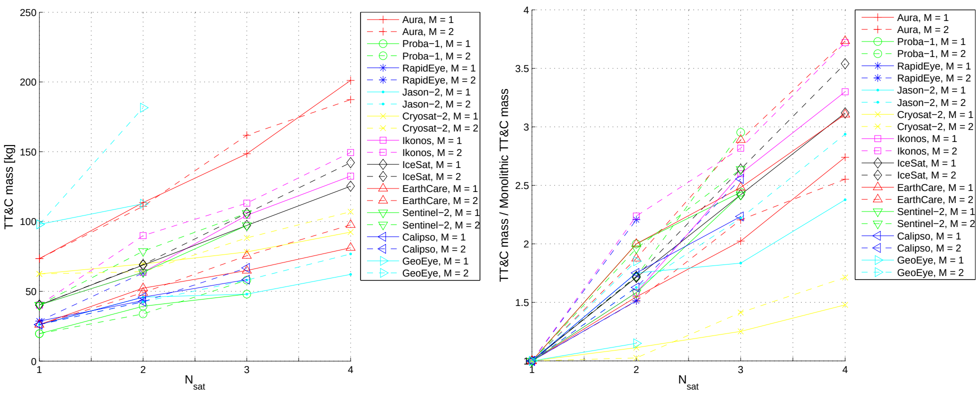


Figure 4.13: Satellites TT&C systems total mass as a function of different configurations

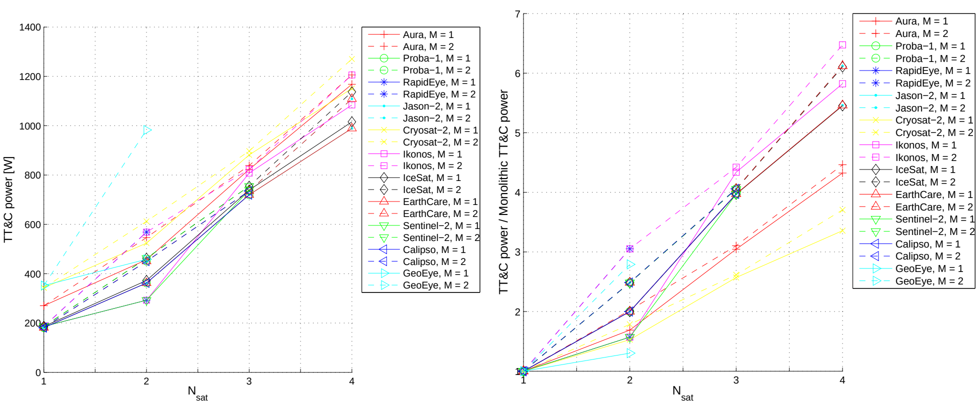


Figure 4.14: Satellites TT&C systems total power as a function of different configurations

cording to possible fractionation; the missing results have been caused by the design algorithm incapacity to find a solution for the proposed configuration. OBDH fraction

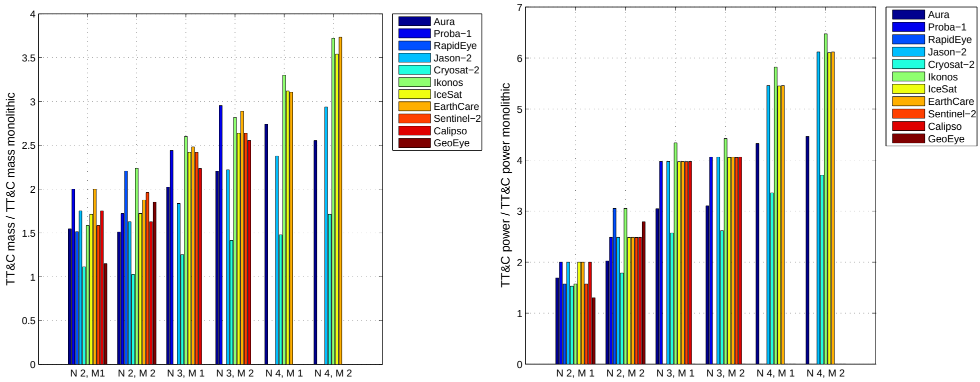


Figure 4.15: Satellites TT&C systems as a function of different configurations

is strictly related to the fulfilment of the local communication network; the impact of fractionation has been analysed both with the combination of the two subsystems, Fig. 4.16 and 4.17, and with an hypothetical remote DH without TT&C, Fig. 4.19 and 4.20. Unlike the communication, the fractionation of the data handling results in a subsystem mass increment with different rates according the number of satellites; the main reason is that the used database for computer components is limited, causing the assembly procedure to reuse the same elements on every satellite. The result is that the complete subsystem for an N_4 spacecraft system will be (or really close to) 4 times the nominal mass.

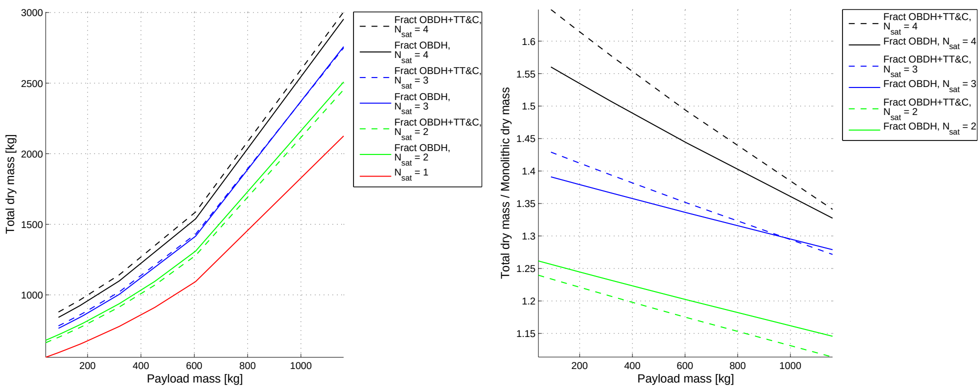


Figure 4.16: Satellites total mass as a function of nominal payload mass and OBDH configuration

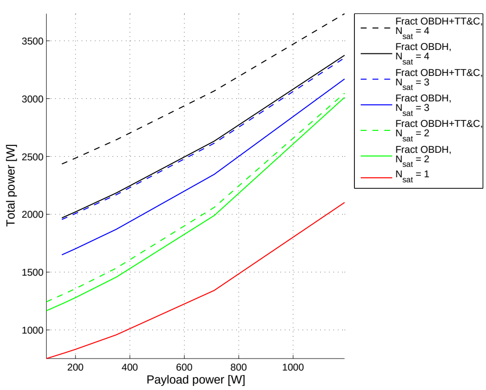


Figure 4.17: Satellites total power as a function of nominal payload power and OBDH configuration

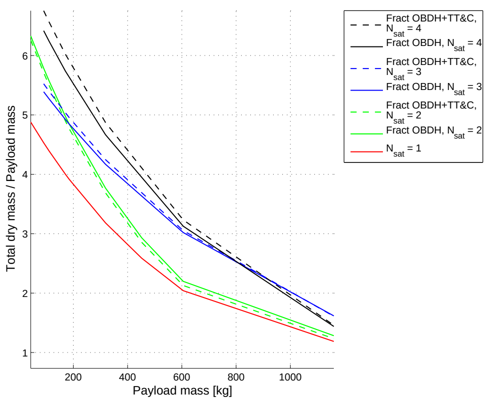


Figure 4.18: Satellite total mass and power increments from monolithic spacecraft according to different OBDH configurations

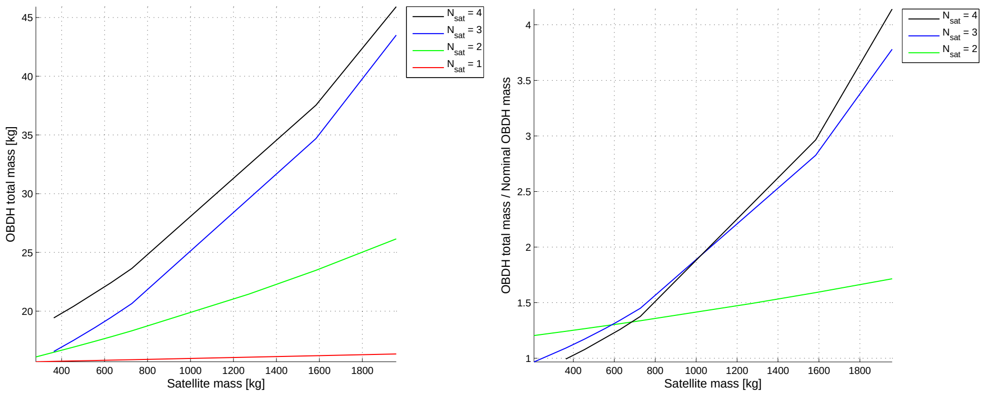


Figure 4.19: Satellites OBDH systems total mass as a function of different configurations

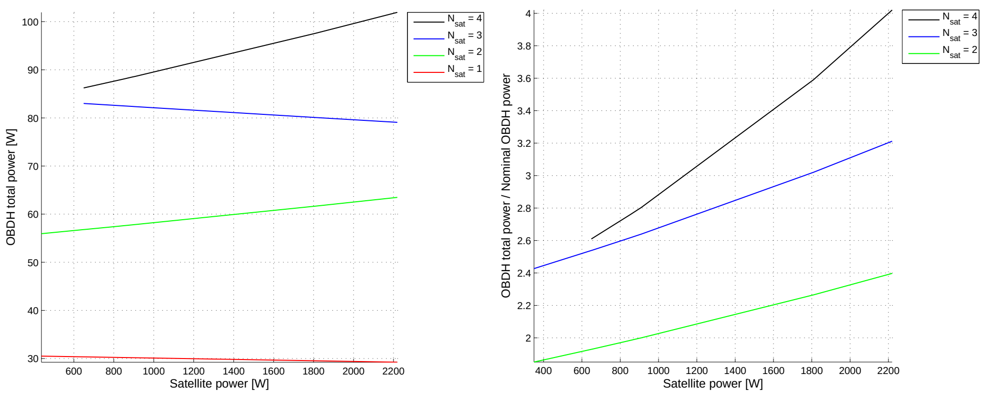
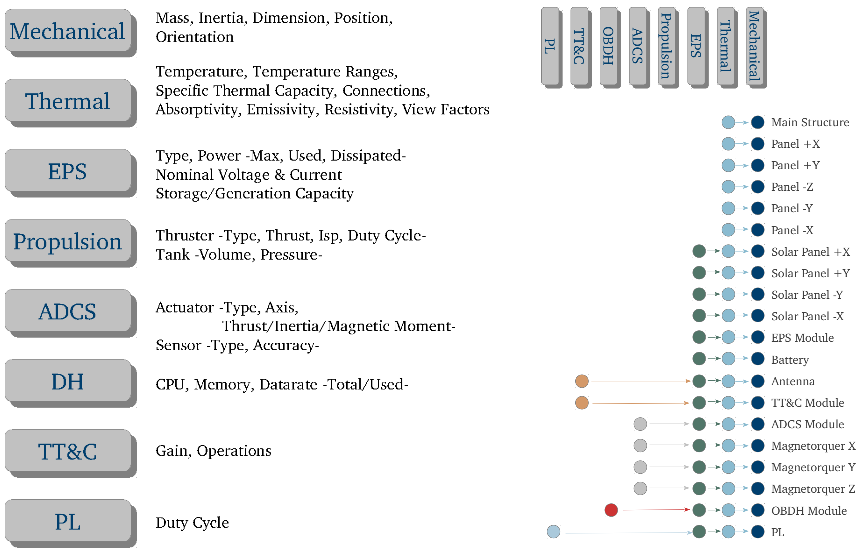


Figure 4.20: Satellites OBDH systems total power as a function of different configurations

CHAPTER 5

Single Spacecraft Operation

A *model* is an abstraction of a real world construct [SC95]. The used methodology involves the development of models for the different elements (i.e. orbital and attitude dynamics, payload physics, subsystems components) of a fractionated satellite system in order to estimate what resources are available at any moment considering previous satellite operations as well as influences due to environment. Although satellite simulators are not a novelty (as has been outlined in Sec. 1.3), most of the existing ones are dedicated to a single system (orbital control) or functionally-coupled systems (attitude and orbit control). Thermal analysis and electric components are commonly addressed with COTS software whereas payloads have specifically-developed dedicated models. The reliability and accuracy level of such tools is often outstanding, but usually they neglect the fall-outs induced by one subsystem on the others, thus requiring the user to iteratively update the subsystem parameters in one tool using the evaluations given by another one. As the number of used satellites increase, this procedure easily become unfeasible, especially if the main task of the simulation is to test inter-satellite interactions. The environment where the satellite operates has a major influence both on design and on operation; as the intention is to test as realistically as possible the system performances a detailed model of the Low Earth Orbit is required. This allows the evaluation of orbital and attitude disturbances -Earth's oblateness and irregular mass distribution, rarefied atmosphere, solar pressure- as well as available solar power -planetary ephemeris- and communication windows with one or more selected ground stations. In a distribute satellite tasks breakdown and resources sharing are a key feature; in order to achieve these objectives, several steps have to be completed. At first a correct estimation of the satellite status and available resources is required. There are two concurrent requirements that affect this aspect of the simulation: accuracy and flexibility. The modelled satellite is going to be the sum of its subsystems, thus the greater the accuracy in every model, the greater the fidelity of the resulting spacecraft; on the other hand a model that breaks-down every component of the satellite to its constitutive elements -i.e. divide a thruster into combustion chamber, feeding valves, valves switches, temperature sensor, nozzle, etc.- would lose flexibility and increase the time to assembly and validate the model. Components breakdown is made to test the performances and the design of the component itself but this level of detail is rarely employed in subsystem models [PF98]. Every subsystem is represented by means of elements characterised by numerical features; number and type of the properties depend on the class of component. This means that a component is regarded as a black-box characterised by mechanical, electrical, thermal features with supplementary high-level attributes that are related to the every subsystems -i.e. an engine has specific impulse and thrust whereas an antenna is characterised by gain and beam-width-. A node-based layered approach has been adopted, as briefly displayed in Fig. 5.1-a. The main idea is to divide the component features according to their functional influence on the different subsystems: e.g. the mass could be considered a "mechanical" property as it influences the physical (intended as classical mechanics) behaviour. A more generic mechanical group has been included as structural dynamics has not been included in the simulation, attitude and orbit considers only rigid body motion. Clearly this classification is arbitrary and commonly a property has influences that cannot be restricted to a single subsystem. As an example heat capacity is an extensive property of matter, meaning that is proportional to the size of the system, thus mass influence can be detected also during thermal analysis. When a single feature had multiple implications, the one that required no additional information to affect a system has been assumed as part of the main layer. With respect to the previous example, classical mechanics is directly ruled by the mass of object, whereas heat capacity is the product of mass and specific heat capacity (a property that is "owned" by the thermal subsystem). Every set of features defines a layer; the complete -within the adopted simplification limits-description of a component is given by the merging of the different layers. Information exchange between layers is possible through a direct inheritance sequence; this allows to keep track of the effect of the variations of the component status through the subsystems. An example of such inheritance/layered structure is shown in Fig. 5.1-b. The hierarchy automatically implies that features must be attached coherently: a transmitter in order to work uses power (thus requiring the electrical properties) with non-perfect efficiency, causing dissipation that modifies its temperature as well as those of the nearby components (thermal layer); the thermal effects could be evaluated only knowing mass and configuration of the parts that constitute the satellite (mechanics). Electronic boards, at least one for each of the subsystems are used to diffuse orders as well as to gather telemetry, Fig 5.2-a. In order to coordinate satellite operations, a hierarchical planning system based on the the Autonomous Robot Architecture (AuRA) [Ark87] concept has been adopted. High-level commands are



(a) (b)

Figure 5.1: Distribution of component features over different subsystems (a) and nodes hierarchy (b)

decomposed into simpler instructions that are divided among the interested subsystems according to a heuristic rule-based system [MA95]. In order to evaluate the performances of the spacecraft, a study conducted over the subsystems one by one would lack in consistency; this is because subsystems and components conditions could change over time even when a specific element is not active. As important as the subsystem models, the cross-connections among them need to be considered. In some cases these induced variations could be foreseen and controlled in order to keep them within an acceptable range (as its done for the thermal control). Other -i.e. satellite mass and inertia variations due to fuel consumption- can be addressed by modifying the parameters in the control system -ADCS gains-. Things start to get complicated when the distance between the source of the variation(s) and the measurable effects spread over more than one subsystem or to components non directly involved in the original chain of events. This may be the case of the activation of the payload, commanded by the on-board computer: the operation requires both EPS and OBDH resources, whose exploit introduces additional loads for the involved distribution board and (possibly) batteries. The dissipated energy generates heat fluxes that could affect batteries or solar panel (as well as other components) efficiency. The consequences of the payload activation also depend on satellite attitude and configuration. A simplified scheme of the modelled connections and relationships is shown in Fig. 5.2-b (some connections are not shown for clarity). Another aspect worth to be

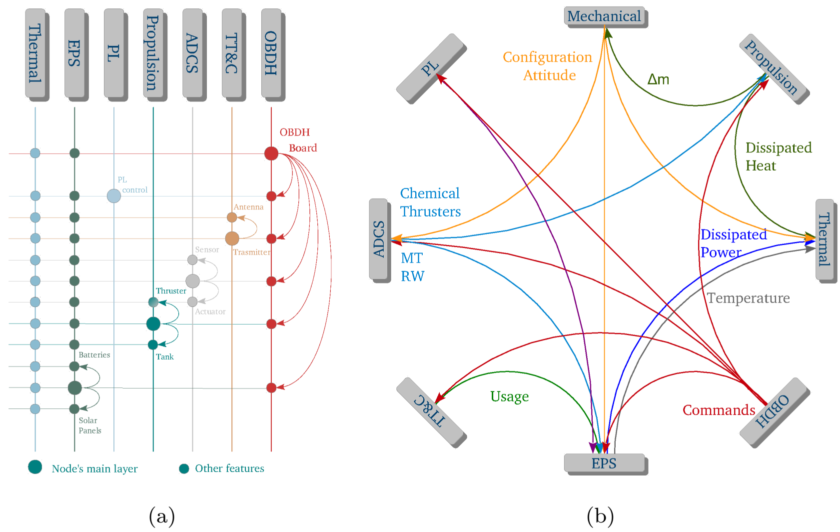


Figure 5.2: Propagation of information through different layers (a) and connections among subsystems (b)

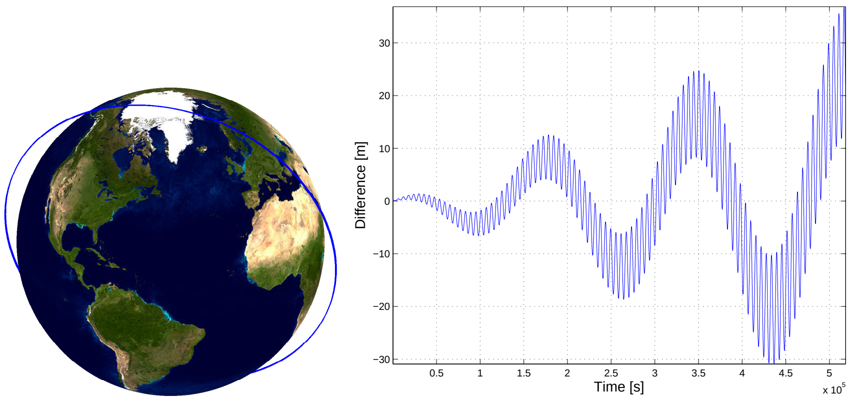
mentioned is that most of the operations need a variable number of pre-conditions, usually related to several subsystems. Than, in order to check if the spacecraft is able to perform a specified task, the status of the whole vehicle should be tested. Among the available integrated simulation tools, just a few are designed to evaluate the side-effects between different subsystems and none of them is specifically addressed to multiple satellites; this lead to the decision to develop a new tool with both single and cooperative spacecraft operation capacity.

5.1 Environment

As modelled satellites are dedicated to Earth observation, an environmental model of the low and mid Earth orbits have been integrated within the simulation tool in order to evaluate attitude and orbit disturbances as well as thermal conditions. Included features are:

- Gravity potential up to J_4 [VAl04]
- Atmospheric drag, using NRLMSISE-00 model for density evaluation [PHDA01]
- Solar radiation [Wie08]
- Geomagnetic field model, using World Magnetic Model (2010) [NOA]

Lunar influence (third body) at the selected orbital range can be neglected [VAl04]. The positions of Sun and Earth with respect to the satellite(s) are also fundamental for solar panels and thermal radiator outcome. Special perturbations method has been used to evaluate the effect of the disturbances and control actions on orbit evolution, as although they are the simplest and the most straightforward of the perturbation methods, they also produce accurate results [VAl04]. Implemented orbital model has been compared with STK software achieving comparable results, Fig 5.3. On a 3 days simulation the difference between the implemented propagator and the SGP4 model is in the order of 40 meters, accurate enough for the tasks demanded to the simulator. In order to ensure orbital changes as well as maintenance, algorithms for orbital



(a) (b)

Figure 5.3: Orbit propagation

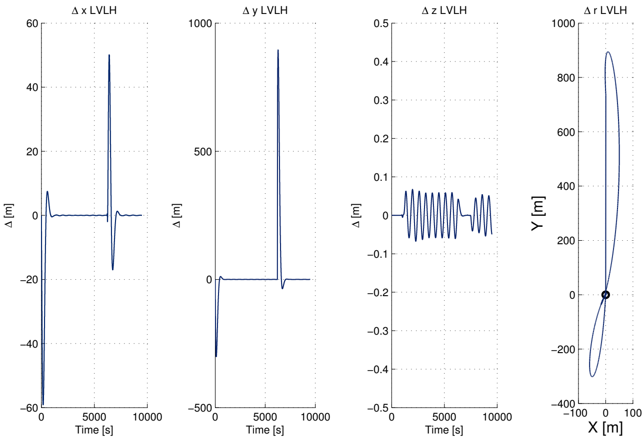
control have been implemented; as for the attitude control, Lyapunov energy-based control functions have been used [SJ03]. The response to a two impulse manoeuvre performed to change the initial orbit has been reported, using both differences between current and reference orbits, Fig. 5.4-a and total distance 5.4-b. The presence of the propulsion system with enough propellant is mandatory in order to execute this kind of manners. The attitude control will provide for the satellite pointing in the required direction during the thrust phases.

5.2 Spacecraft Subsystems

In the broadest sense the satellite itself is only an element within a larger system that includes the ground support and the launcher, but so far the focus has been kept on the spacecraft.

5.2.1 Payload

Although technically not a subsystem, is the fundamental reason the spacecraft is flown. Especially for science-dedicated satellites the simulation and validation of the



(a) (b)

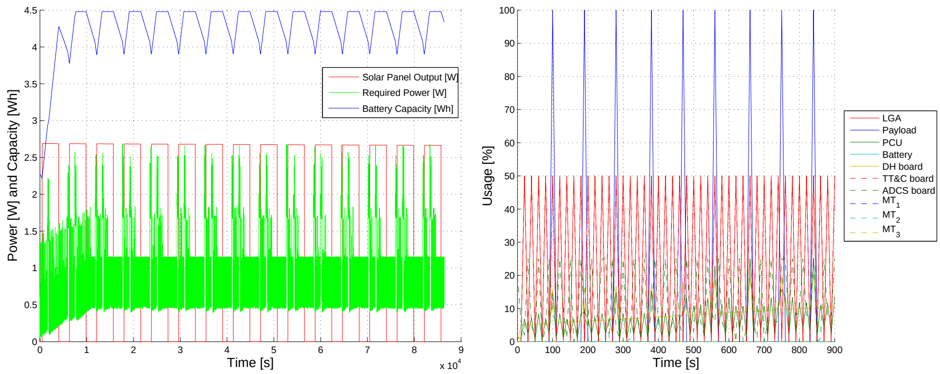
Figure 5.4: Orbit control

on-board instruments can be extremely complex but for the perspective of the current work, they can be roughly approximated as resources-user black-boxes. By knowing the amount of power needed both during operation and stand-by phases, the generated data-rate and the duty cycle its possible to evaluate the impact of the payloads on the overall satellite capacities. Payloads are commanded by the C&DH and activated when pre-conditions (duty cycle, available power, lightning conditions) are satisfied.

5.2.2 Electrical Power System

The EPS has the function to provide, store, distribute and control the electrical power within the spacecraft. The EPS module is a simple generation-distribution-storage model that takes into account power usage, production and losses connected to the distribution [LCKL88]. The main elements modelled in the tool are:

- Solar Arrays $f(attitude, temperature)$
- Batteries $f(temperature)$
- Power Conditioning Unit
- Laser Source and collimation/aiming mirror
- Microwave generator /focusing antenna
- Rectenna (RF/DC converter)
- Loads (payload, electronic components, actuators, etc.) $f(status)$



(a) (b)

Figure 5.5: EPS evolution (a) and nodes absorbed power (b) simulation

$f()$ has been used to highlight the dependence of a class of components from effects not modelled in the EPS framework. This could be a reduction/increase in efficiency (solar panel) or a variation in the used power (loads, as a result in their activation commanded by the OBDH). The aim is to evaluate the evolution of:

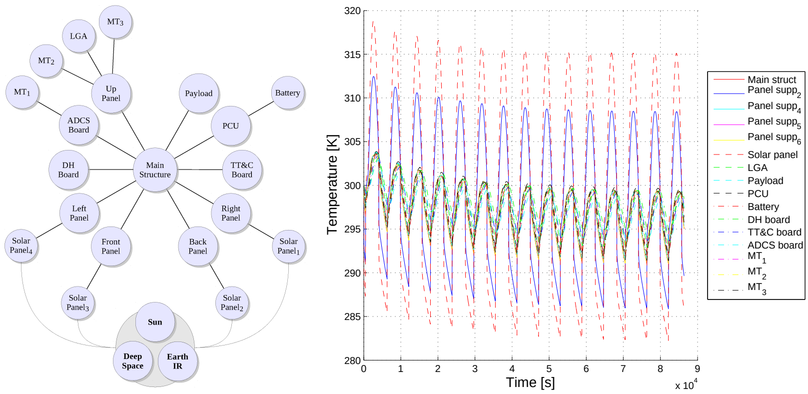
- available power (generated autonomously or transferred)
- battery state-of-charge
- used power
- dissipated power

A typical output of the EPS simulation is reported in Fig. 5.5. The sharp variations in produced power are due to periodical eclipses.

5.2.3 Thermal

The control of the temperature of the spacecraft equipment and structural elements is required for two reasons [FSS11]: most of electronic and mechanical parts are designed to work efficiently within a narrow temperature range and materials typically have a non-zero thermal expansion coefficient thus meaning that a temperature change would introduce a thermal distortion. Active or purely passive controls can be used, as well as combination of them. The spacecraft has been represented using a lumped system analysis [IDBL07]; the satellite is reduced to a network of discrete spots connected by thermal resistances (conductive and radiative). In fact by applying the principle of the conservation of energy to every node of the system:

$$Q_{in} - Q_{out} = \Delta Q_{stored} \tag{5.1}$$



(a) (b)

Figure 5.6: Thermal network

where Q_{in} and Q_{out} are the inward and outward heat flux respectively, Q_{stored} is the amount of stored power, and considering both Fourier's law, (Eq. 5.2) and Stefan-Boltzmann's law for a gray body (Eq. 5.3), system can be written as Eq. 5.4 for $i, j = 1 : n$ [IDBL07].

$$Q = -K\,A\,\Delta T \tag{5.2}$$

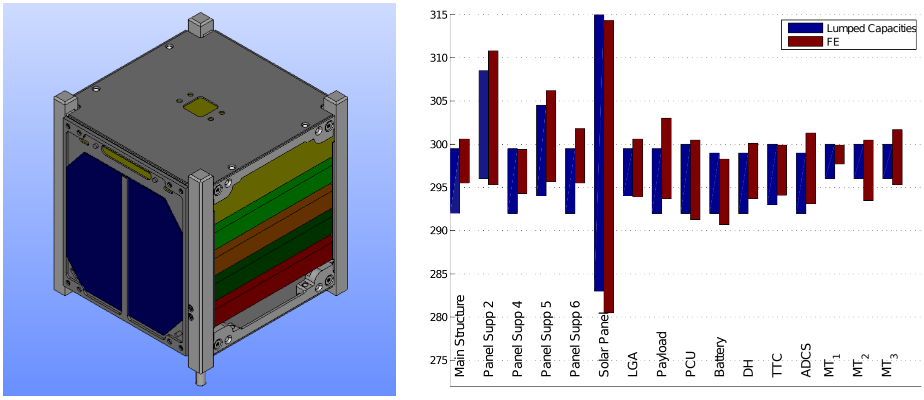
$$Q = \sigma\,\alpha\,\epsilon\,F\,A\,\Delta T^4 \tag{5.3}$$

$$\sum_{i \neq j} K_{ij}\,(T_j - T_i) + \sum_{i \neq j} R_{ij}\left(T_j^4 - T_i^4\right) + Q_i = C_i \frac{dT_i}{dt} \tag{5.4}$$

K_{ij} represents the thermal resistance (conductive) between nodes i and j, R_{ij} the radiative resistance and C_i the heat capacity of node i. Thanks to this approximation is possible to simplify otherwise complex differential heat equations. The discretised set of equations has been integrated using a first order, implicit method (backward Euler). Heat contributions to each nodes can be both internal and external:

- Dissipated electrical power
- Dissipated thermal power (i.e. thrusters)
- Heat flux through conduction
- Radiative heat flux to or from external sources

The first two inputs are strictly related to the operative status, whereas the other two depend on position, orientation and type of connection of the element. At the state-of-the-art of the tool, the control of the components temperature is achieved using thermal coatings, radiators and heaters. Nodes network, Fig. 5.6 is assembled using information about components mass, specific heat and geometry; thermal resistances have to be specified by the user as their evaluation, is extremely difficult and could hardly be made using a simplified model [Gil02]. The calculated temperature



(a) (b)

Figure 5.7: CAD model used for the thermal analysis and results comparison

profile for the same simulation shown for the EPS is reported in Fig. 5.6-b. The thermal analysis has been tested (same nodes properties, connections and boundary conditions) using the Femap Thermal Solver on a 3d model of the satellite, Fig. 5.7. Minimum and maximum temperatures, achieved during the 8th orbit, evaluated using both the implemented lumped capacities and the validation FE model have been reported in Fig. 5.7-b (for FE the average component temperature has been used).

5.2.4 Attitude determination and Control

The attitude determination and control system provides stabilisation for the spacecraft and orients it in the desired direction during the mission, in spite of external disturbances. In order to achieve this goal, two main tasks have to be performed: attitude determination using sensors -and status estimators- and control [LW99]. Currently, sensors are included in the mass and power budgets but they do not have a mathematical model within the ADS abstraction i.e. attitude, position and velocity are known and error-free. The complete spacecraft dynamics is calculated by integration of the Newton-Euler equations [Wie08]. Control torques determination is achieved thanks to Lyapunov Control Functions [SJ03]; this technique has been chosen over a more conventional Proportional Integral Derivative control to better assess variations in the dynamical model and guarantee an higher controllability in case of manoeuvres involving large angles [BLH01]. Control torques are then processed according to the class of used actuators in order to calculate required power and -if needed- propellant to perform the manoeuvre. Maximum and minimum applicable torques are self-computed given the features of the control devices, their position within the spacecraft and the orbital parameters (as magneto-torquers are influenced by the Earth's magnetic field).

5.2.5 On Board Data Handling

The C&DH receives, validates, decodes, and distributes commands to the subsystems and at the same time gathers, processes, and formats satellite housekeeping and mission data for down-link or use by an on board computer. In order to manage the satellite, at least a simple control architecture has to been implemented; it has to fulfil several objectives [SMR]:

- *Programmability:* satellites cannot be designed for only one precise task. They should be able to achieve multiple tasks which should be described at some abstraction level. The functions should be easily combined according to the task to be executed.
- *Reactivity:* spacecraft should be able to take into account external events with time bounds that are compatible with the correct and efficient execution of their task (including their own safety).
- *Autonomy and adaptivity:* satellites are designed to accomplish their task by themselves despite of possibly changing external circumstances.
- *Robustness:* the control architecture should be able to exploit the redundancy of the processing functions (for example by decentralising all parts of the control system that does not explicitly require centralisation).
- *Coherent behaviour:* to achieve the main task, reactions of the spacecraft to external stimuli must be guided by the objectives of the task.
- *Observability:* to be controllable at higher level, the functional level should make accessible a representation of its activity and data concerning the state of the rover or information about its environment.

In order to achieve these purposes a hybrid deliberative/reactive robotic architecture that was developed in the mid-1980's and known as AuRA has been integrated with the simulated OBDH. The original code [Ark87] was a hybrid approach to robotic navigation. Hybridisation arises from the presence of two distinct components: a deliberative or hierarchical planner, based on traditional artificial intelligence techniques; and a reactive controller, based upon schema theory [FC94]. A hybrid system must combine the two time-scales and representations of the deliberative and reactive architectures in an effective way. Usually, a middle layer is required. Its mission is to achieve the right compromise between the two ends. Consequently hybrid systems are often called *Three-layer-architectures*. The highest level is concerned with establishing high level goals; in this case it has been simulated by loading a sequence of tasks (equivalent to send instructions to the spacecraft from the control centre). Issued commands are stored and executed when pre-conditions are met, e.g. downlinks can be established only when a communication window is open. The main instructions are then broken down into sub-tasks and distributed among the relative subsystems where electronic boards are responsible for their control and execution. The implemented concept connects two distinct AI paradigms, deliberation and reactivity; the advantages of this strategy have been demonstrated in several other hybrid architectures that have subsequently appeared [GKBM98]. The disadvantages of this architecture arise from the presence of different modules: it inherits the drawbacks on the single modules plus the complexity of the creation of an harmonic link between them [KS03]. Although old-fashioned, the used three layers architecture has been selected considering its straightforward implementation and simplicity compared with modern (more capable but also more complicated) approaches. Apart from the management of the spacecraft, the DH also has to access, retrieve and store science and telemetry information. The adopted model for these functions is extremely simple, basically a conservation equation: data movements cause variation in the used memory as they come to (routed from instruments, communication receivers or components telemetry) or they leave the satellite (download to Earth or another spacecraft), Fig. 5.8. At the same time, the absorbed power by the involved components is updated.

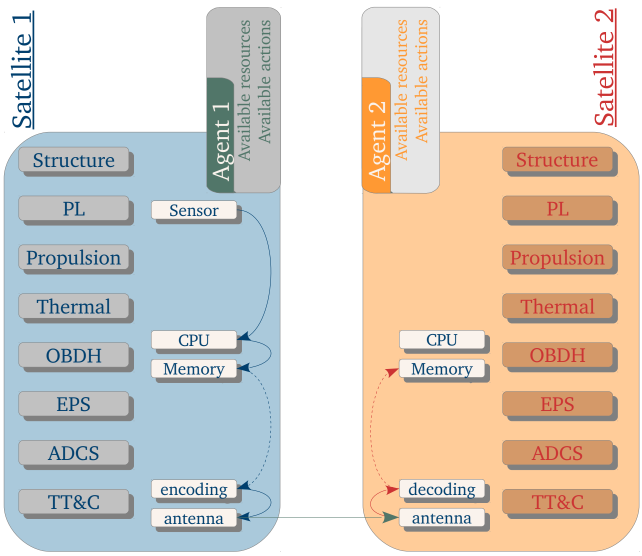


Figure 5.8: Data transfer among 2 satellites

5.2.6 Telecommunications

The TT&C has not a model in the mathematical sense; it works as a bridge between the simulated ground stations (or satellites) and is responsible for the evaluation of the incoming (or outgoing) datarate. Its real functions like tracking, encoding/decoding, compression and analysis where beyond the task of the present work; when communication devices are activated by the OBDH, the data flux in enabled and the absorbed/dissipated power are updated coherently.

5.2.7 Propulsion

The Propulsion subsystems has been sketched with thrusters, propellant and (if any) pressuriser tanks. The activation of the attitude or orbital engines causes heat dissipation, absorption of electric power and a reduction of the remaining propellant mass. Once again, a conservation equation holds the model of the subsystem. Fuel consumption is evaluated according to thrusters specific impulse, mass ratio and duration of the activation. Mass, inertia and pressure of the propellant inside the tank(s) is updated at each time-step. Thruster model is simplified, and upon activation the maximum thrust is applied without transient phase.

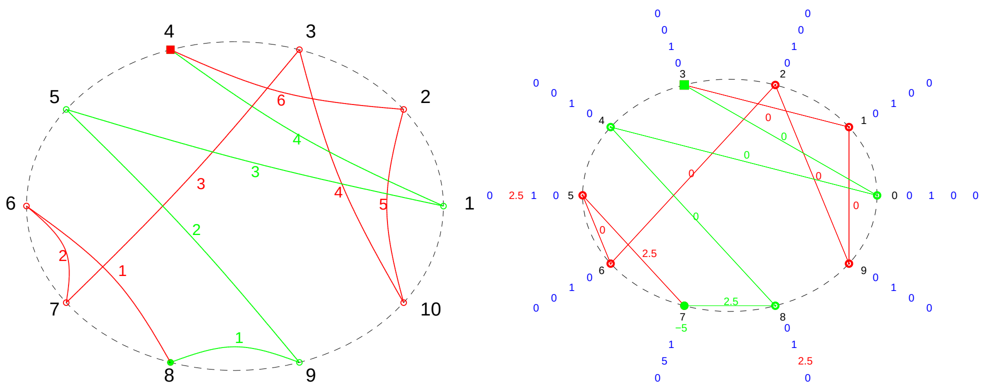
CHAPTER 6

Spacecraft System Operation

At the state of the art, only communication-type inter-satellites connections have been modelled: even if the adopted management system could be also extended to power transmission, it is an option that is always discarded during design optimisation, thus its implementation has been postponed. The NASA's TDRS is an example of working satellites connection system; from a general point of view it is not different from the concept of spacecraft-ground station communication, the main difference that lies in the fact that the TDRSS elements are moving to. Communications between satellites has several advantages [CGG13], mainly related to the low -compared with the atmospheric propagation- free space losses that allows smaller, lighter and less power demanding systems. Nonetheless within the spacecraft system, at least one element with a traditional high gain antenna is still required.

6.1 Game Theory-Based Satellite Communications

A game theory-based control of the data sharing among satellites, partially based on [FHB06] has been implemented. Each node (satellite) is connected to one or more of the surrounding nodes; the connectivity is represented by means of a directional graph, Fig. 6.1-a. Single hop connections allow communications between node i and $i+1$ lying on the same path. Communication between non-neighbouring nodes is based on multi-hop relaying, thus forcing the source node to rely on intermediate nodes to deliver his data to the end node. The list of nodes (vertex of the graph) and links that connect source and destination form a path; the time required to deliver an information package (or fraction of it) depends on the connectivity graph (assuming intermediate nodes cooperation). Data packages are assumed to be divisible; if a single node is shared among several paths the throughput will be equally divided among the outgoing routes, Fig. 6.1-b (e.g. node 7 generates 5 units of traffic that are routed through 2 paths, 2.5 units each). Packages symbolise output from scientific instruments, telemetry and commands that have to be transferred to and from a satellite able to communicate with the ground station (thus acting as comms hub).



(a) (b)

Figure 6.1: Randomly generated Networks (The information placed radially with respect to a node are ID, payoff, cooperation, throughput and used memory)

6.1.1 Forwarding Game

The operation of the network has been modelled as a game, whose structure is similar to the forwarder dilemma [Mye97]; each node must pay (representation of power consumed by the TT&C) in order to send the package along the path but only the source receives a reward for having his own data delivered. So the players (the nodes) at each time-step t choose a cooperation level $p_i(t) \in [0, 1]$; 1 represents full cooperation, meaning that all the received packages are forwarded whereas 0 is associated to full defection (no data are forwarded). Thus, $p_i(t)$ represents the fraction of the traffic routed through i in t that i actually forwards. Non-forwarding has been considered as the only reason that could lead to data loss (i.e. no network congestion, power or data rate limits etc.). Later on, modifications on $p_i(t)$ meaning and values will be introduced. In order to simplify the model, the same level of cooperation is applied from node i to every paths it is linked to. The generic route r connects the source s and destination nodes through $f_1; f_2; \cdots ; f_l$ the intermediate points. $T_s(r)$ is the traffic that s wants to send on r in each time-step. The throughput $\tau(r, t)$ experienced by the source s on r in t is defined as the fraction of the traffic sent by s on r in t that is effectively delivered to the destination. Hence, $\tau(r, t)$ can be computed as the product of $T_s(r, t)$ and the cooperation levels of all intermediate nodes (with no re-routed packages):

$$\tau(r,t) = T_s(r) \cdot \prod_{k=1}^{l} p_{f_k}(t) \tag{6.1}$$

Re-routing can happen when multiple paths have access to the same nodes in a different sequence. In the original paper [FHB06], a normalised form of the throughput has been defined and used as input for the evaluation of the strategy; due to the introduction of the shared nodes and the subsequent data division among several paths, that approach has been dropped and the effective throughput has been adopted in order to evaluate with an higher fidelity costs and gains. The Payoff (PO) $\xi_s(r,t)$ of s on r in t depends on the experienced throughput $\tau(r,t)$. The payoff $\eta_{f_j}(r,t)$ of the j-th intermediate node f_j on r in t is non-positive and represents the cost for node f_j to forward packets on route r during time step t. It is defined as:

$$\eta_{f_j}(r,t) = -c_f \cdot \tau_j(r,t) \tag{6.2}$$

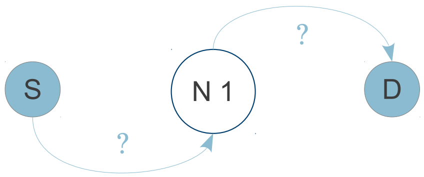
$\tau_j(r,t)$ is the amount of traffic that is passing through node j (in the original statement the normalised throughput has been used). If a single node is shared among N_j paths, its total forward cost is given by the sum of the packages send over the differ paths:

$$\eta_{f_j}(t) = \sum_{k=1}^{N_j} -c_f \cdot \tau_j(k,t) \tag{6.3}$$

c_f depicts the power consumed by the satellite to forward one unit of traffic (either generated by the spacecraft itself or relayed). An introduced, although unsophisticated hypothesis, is that the transmission cost is the same for all nodes, regardless for the distance between them, and constant over time; this is based on the assumption that inter-satellite distance is controlled and with regulated range variations (thus leading to nearly constant free-space losses). Using this approach, the payoff of the destination node is equal to zero, due to the fact that only the nodes that generate the data flow benefit if the their packages are delivered. The total payoff $\pi_i(t)$ of node i in time slot t is computed as the sum of internal ($\xi(r,t)$) and external ($\eta(r,t)$) payoffs.

$$\pi_i(t) = \sum_{q \in S_i(t)} \xi_i(q,t) + \sum_{r \in F_i(t)} \eta_i(r,t) \tag{6.4}$$

The method used to evaluate the payoff causes the game to be slightly different from the classical forwarder's dilemma [Mye97] but leads to the same result, the only Nash Equilibrium (NE) of the game is the non-cooperation of the nodes. This is related to the initial statement that only source nodes can expect a payoff larger than zero, whereas all the others will relay information (thus consuming their own resources) to gain nothing. Assuming a single package, payoff $\eta_{f_j}(r,t)$ and cost c take the values of 1 and c_f respectively. Using iterative weak dominance results that



		$\mathbf{N_1}$	
		F	D
S	F	$1 - c_f,\ -c_f$	$-c_f\ 0$
	D	0, 0	0, 0

Figure 6.2: Strategic form for 2 players, S and N_1

N_1 will drop the package, as this strategy gives a minimum payoff of 0 regardless for S action [Mye97, Str93]; then S will choose not to forward the package, to avoid a loss equal to c_f. The final result is that no packages will be delivered. Several solutions to incentive nodes cooperation has been proposed, either by denying service to misbehaving nodes through a reputation mechanism, increasing the payoff of the altruist players or introducing penalties related to lost packages.
The time required to forward the data from source to destination along a path is greater than a single simulation step; this has a non negligible effect on the chosen cooperation level of the nodes, introducing a delay between the change in strategy and the variation in terms of expected payoff. In particular, even starting from a nominal condition with full cooperating nodes the payoff is negative for all the elements (including the source), due to the fact that no packages are reaching the destination as they are still en route. Then without any incentive strategy-evaluations based on the payoff at the last time-step will result in the stop of the data forwarding.

6.1.2 Memory and Storage Cost

In the conventional form of the game, non-forwarded packages are lost. Considering that the packages typically represent valuable data, this is non-desirable, and the exploit of the on-board memory to store a limited amount of information has been introduced; non-forwarded packages are progressively recorded up to the maximum allowed. Related to the memory capacity, a cost for data storage has been added, c_s, proportional to the percentage of use. The insertion of the memory naturally results in a penalty for non-cooperating spacecraft; as the node does not forward the packages it has to pay an increasingly larger price for their storage, up to the point where the data transmission is preferable. The intermediate node payoff function has been modified accordingly, for non-shared node:

$$\eta_{f_j}(r,t) = -c_f \cdot \tau_j(r,t) - c_s \cdot memory_{used_j} \tag{6.5}$$

And consequently for multiple-paths nodes:

$$\eta_{f_j}(t) = \sum_{k=1}^{N_j} (-c_f \cdot \tau_j(k,t)) - c_s \cdot memory_{used_j} \tag{6.6}$$

In order to represent an incentive to data transmission, storage cost must be scaled considering to available memory, expected node throughput and forward cost. The node strategy at time t is determined using the information about used memory and throughput at time $t-1$; the cooperation level is calculated as the weighted sum of the percentage of used resources scaled by the cost factors, Eq. 6.8.

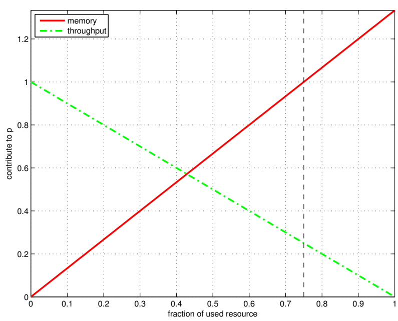
$$\begin{aligned} \mathsf{mem} &= \frac{memory_{used}}{memory_{available}} \\ \mathsf{th} &= \frac{throughput_{used}}{throughput_{available}} \end{aligned} \tag{6.7}$$

$$p(t) = \frac{p_{mem} \cdot \mathsf{mem} + p_{th} \cdot \mathsf{th}}{p_{mem_{max}} \cdot \mathsf{mem} + p_{th_{max}} \cdot \mathsf{th}} \tag{6.8}$$

p_{mem} and p_{th} are evaluated using linear equations:

$$\begin{aligned} p_{mem} &= a_{mem} \cdot \mathsf{mem} + b_{mem} \\ p_{th} &= a_{th} \cdot \mathsf{th} + b_{th} \end{aligned} \tag{6.9}$$

Shape and slope of the functions has been chosen in order to:



	a	b
p_{mem}	$\frac{(cost_s)^{-1}}{intersec\,point}$	0
p_{th}	$-(cost_{th})^{-1}$	$(cost_{th})^{-1}$

Figure 6.3: Cost functions parameters

- reduce cooperation as the throughput approaches the node's maximum data rate, thus avoiding network congestion
- increase cooperation as the node runs out of memory

In this way the node has to choose between cooperation (and the fixed price that has to be paid for package forwarding) and non-cooperation associated to a continuously increasing penalty. The maximum values reached by the curves are controlled by the cost parameters. In order to allow cooperation to assume values larger that 1 the slope of the memory curve has been increased by moving the maximum nominal value to .75 of the used resource (instead of 1.). Cooperation levels greater than the unity implies that the node is routing all of the received packages plus a part of the data stored within its memory. Node's behaviour can be controlled by means of the position of the memory nominal maximum value, thus increasing/decreasing his attitude toward cooperation. The are limits to this control action, as the exploitation of low values for the intersection point leads to proportionally higher maximum real values for p_{mem} that may cause instabilities in the network Fig 6.4.

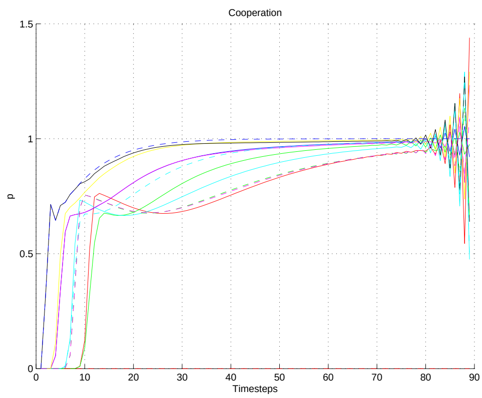


Figure 6.4: Instabilities introduced by overestimated cost function parameters

6.2 Forwarding Game, 12 Nodes and 3 Paths

The features of a randomly generated network imposed of 12 nodes over 3 paths are reported in Tab. 6.1, 6.2, 6.3 and Fig. 6.5. In order to maximise delivered data rate, cooperation has been imposed as well as used as free variable for a PSO-type optimisation algorithm. Payoffs and costs over time for the nodes are plotted in Fig. 6.6. The paths have different lengths and involved nodes but share source and destination (although source sharing has not been imposed).

Input DR	5	$\frac{package}{time-step}$
$cost_f$	1	-
PO function	linear	-

Table 6.1: Input parameters

Path ID	N. vertex	N. links
0	11	10
1	4	3
2	7	6

Table 6.2: Network structure

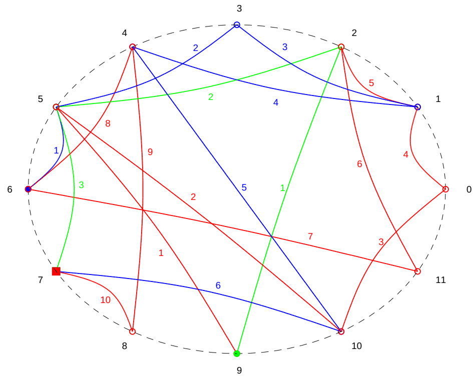


Figure 6.5: Structure

Node	Connections In	Connections Out	DR
0	1	1	0
1	2	2	0
2	2	2	0
3	1	1	0
4	2	2	0
5	3	3	0
6	1	2	5
7	3	0	0
8	1	1	0
9	0	2	5
10	2	2	0
11	1	1	0

Table 6.3: 12 Nodes over 3 paths

6.2.1 Free Storage

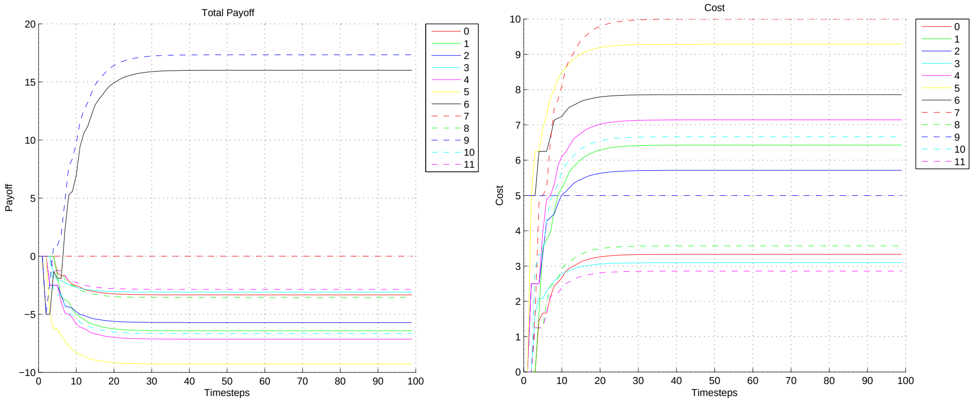
Results for network without memory penalties are shown in Tab. 6.4 and 6.5. The 2 source nodes have a different payoff, due to the fact that all the packages are routed differently. In order to maximise delivered data rate cooperation has been imposed as well as used as free variable for a PSO-type optimisation algorithm. Payoffs, costs and nodal throughput over time for the nodes with forced cooperation are plotted in Fig. 6.6 and 6.7. As expected, players cooperation is not trivial as they get a better payoff (0) for non forwarding the packages (the PSO highlighted that only the behaviour of the source nodes has a relevance, as their non-cooperation results in null payoff for all of the nodes), Tab. 6.5. The best strategy for the nodes is the non-cooperation,.

		0	1	2	3	4	5	6	7	8	9	10	11
Theory	Coop.	1	1	1	1	1	1	1	1	1	1	1	1
	PO	-3.3	-6.4	-5.7	-3.1	-7.1	-9.3	16.	0	-3.6	17.3	-6.7	-2.9
PSO	Coop.	1	1	1	1	1	1	1	1	1	1	1	1
	PO	-3.3	-6.4	-5.7	-3.1	-7.1	-9.3	16.	0	-3.6	17.3	-6.7	-2.9

Table 6.4: Max delivered data rate, total payoff -14.75, PSO population 96 elements, 16 step

6.2.2 Storage Cost

The cost for storing packages has now raised to 0.15 in order to keep the cost for full memory occupation higher than the worst case transmission cost. Evolution of the payoffs, costs and cooperation are shown in Fig. 6.8 and 6.9. This time the evaluation of the foreseen results is complex due to their dependency to the network structure.



(a) (b)

Figure 6.6: Payoff and costs for forcedly cooperating nodes

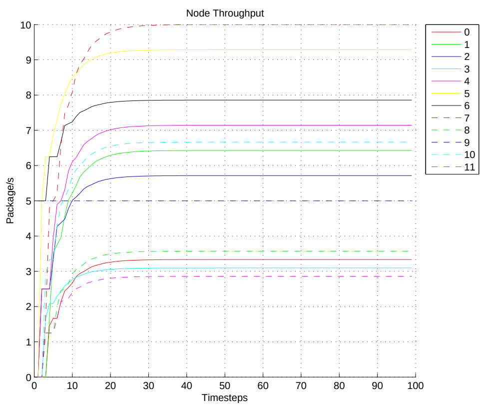
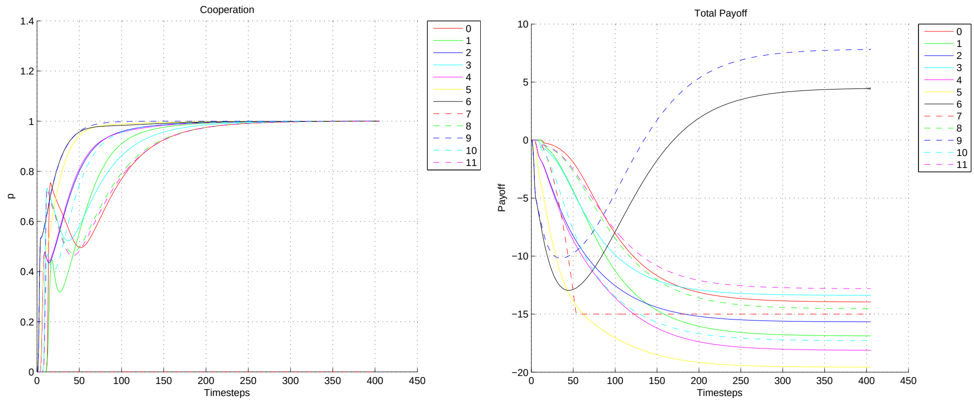


Figure 6.7: Throughput for forcedly cooperating nodes

		0	1	2	3	4	5	6	7	8	9	10	11
Theory	Coop.	-	-	-	-	-	-	0	-	-	0	-	-
	PO	0	0	0	0	0	0	0	0	0	0	0	0
PSO	Coop.	-	-	-	-	-	-	0	-	-	0	-	-
	PO	0	0	0	0	0	0	0	0	0	0	0	0

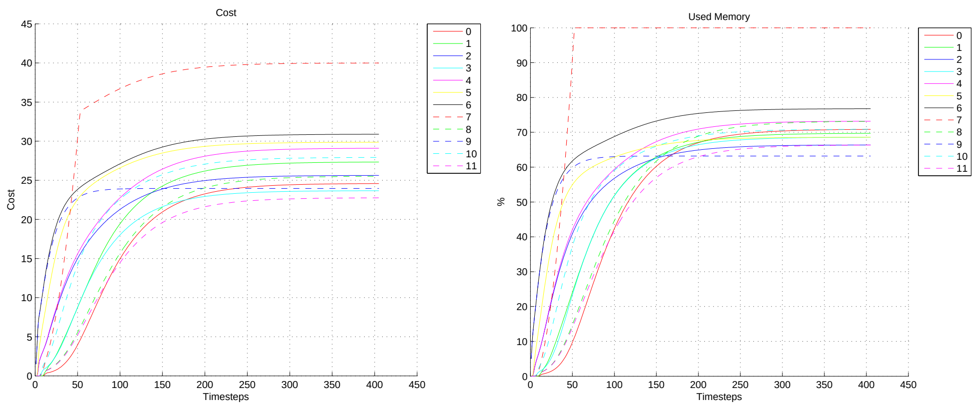
Table 6.5: Max payoff, total 0, PSO population 96 elements, 1 step

The shape of the cooperation curve is caused by the weighted sum used to set the



(a) (b)

Figure 6.8: Cooperation and Payoff with storage cost



(a) (b)

Figure 6.9: Cost and Memory with storage cost

strategy and represents the transition between throughput and memory dominated regimes.

6.3 5 Satellites, 2 Paths

The game has been applied to a simulated networks of 5 satellites with fractionated communication system. Each satellite is composed by models dedicated to its main subsystems, components and their cross-connections as described in the pre-

	0	1	2	3	4	5	6	7	8	9	10	11
Coop.	1.	1.	1.	1.	1.	1.	1.	0	1.	1.	1.	1.
PO	-13.9	-16.9	-15.7	-13.4	-18.1	-19.6	4.5	-15	-14.6	7.7	-17.3	-12.8

Table 6.6: Cooperation and Payoff for time evolving game with storage cost

vious chapter. In this way, every time a component uses a resource, its effects are considered not only at system level; instead their global influence is evaluated.

6.3.1 Satellite Set-Up

A simplified model of spacecraft network has been modelled, using 5 units divided in 2 paths ending with a single comms hub. Simulation time is 8640 seconds, 1/10 of a day, enough to complete 1.5 sun-synchronous orbits. The satellites are based on the same structure; S_4 and S_5 carry scientific payloads, S_1 hosts a dedicated communication downlink system. All the satellites have limited-bandwidth transmitters/receivers to establish spacecraft-to-spacecraft links but have the same electronic board, Tab. 6.7. The differences within initial allocated memory are used only to reduce curves overlapping in the following graphs.

Node	Connections		DR	Total Mem.	Used Mem.
	In	Out	[MBps]	[Mb]	[Mb]
0 (S_1)	2	0	0	2000	100
1 (S_2)	1	1	0	2000	150
2 (S_3)	1	1	0	2000	200
3 (S_4)	0	1	.15	2000	250
4 (S_5)	0	1	.1	2000	300

Table 6.7: Network structure

6.3.2 Imposed cooperation

The evolution of the network has been initially analysed forcing the cooperation level; this is equivalent to allow the satellites to communicate every time they have some data, Fig. 6.11 and 6.12. The information is routed till the comms hub (S_1) and there stored until it the download window opens (the 3 reductions in stored data level). This kind of operations result in S_1 being overloaded whereas the other satellites have plenty of free space (heuristics to control this aspects can be used, like transmit data sat-to-sat only if used memory is greater than i fixed value). Memory level for non-ground-connected satellites is constant as $p = 1$ implies that the output data-rate is equal to the incoming/generated one, thus leaving memory untouched. Consequently

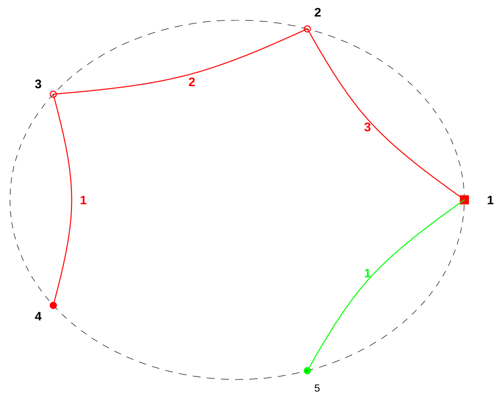


Figure 6.10: Network Scheme

Input DR	.15 (S_4) & .1 (S_5)	Mb/s
$cost_f$	1	-
PO function	$2 \cdot \|throughput\|$	-

Table 6.8: 5 satellites input parameters

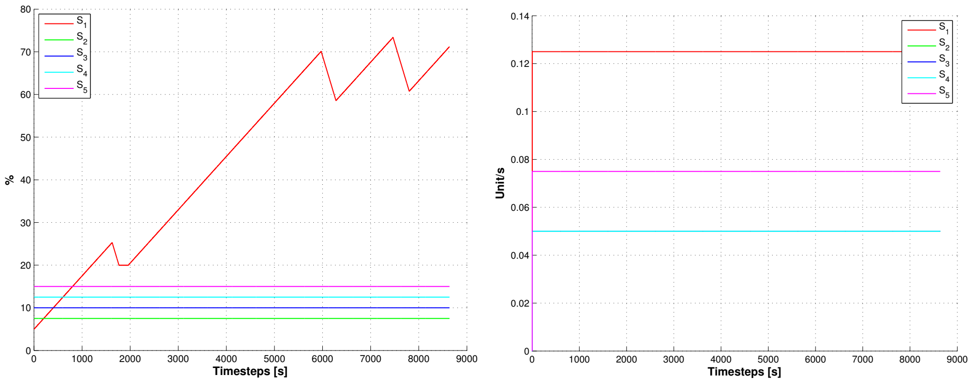
Path ID	N. vertex	N. links
0	4	3
1	2	1

Table 6.9: Structure of the 5 satellites network

also the cost paid by the single nodes is constant. Things change when storage cost is introduced and cooperation is set free, Fig. 6.13, as memory occupation is associated to an increase in cost.

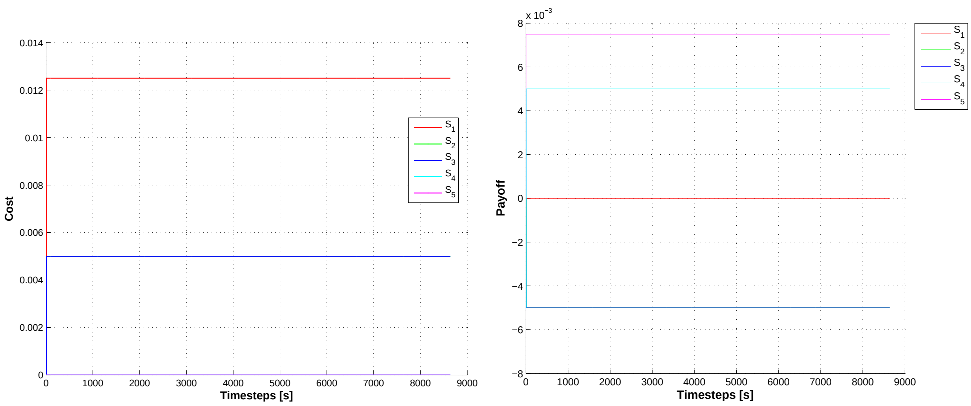
6.3.3 Free cooperation

Simulation time has been increased to 43200 s (half a day); the satellites have the same configuration. Cooperation level is used as control variable, regulated by the transmission and storage costs. Variables evolution over time is reported in Fig. 6.14 and 6.15. Memory and throughput exhibit a smooth behaviour, due to the continuous p variation according to every satellite conditions. Data flow is incremental, with rates that increase up to a steady condition; control parameters of p have been set in order to achieve equality in data collected-transmitted with 60% of memory used. Throughput of S_1 is initially driven by S_5 as they have a direct connection, S_4 influence can be detected only subsequently. Data are divided among the memory of the satellites, thus avoiding S_1 on board hard-disk overuse. The test represents the worst case as the incoming data-rate is constant (whereas in reality science instruments duty cycle will decrease the average data generation) and therefore larger than the comms hub download capacity; it is inevitable that S_1 will run out of free space, but the shared memory allows the system to operate for a considerably longer time (that



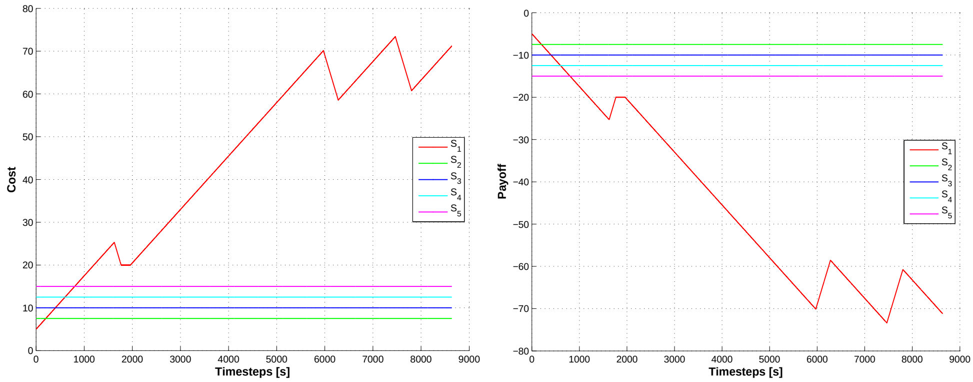
(a) (b)

Figure 6.11: Memory (a) and throughput (b) for $p = 1$, no storage cost



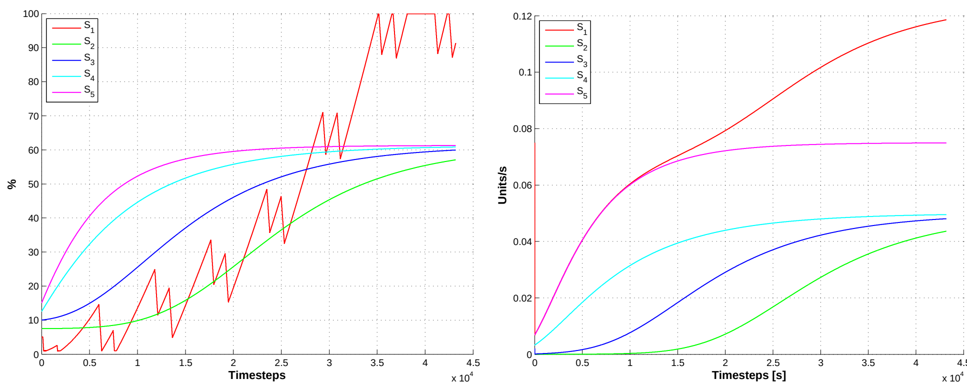
(a) (b)

Figure 6.12: Cost (a) and Payoff (b) for $p = 1$, no storage cost



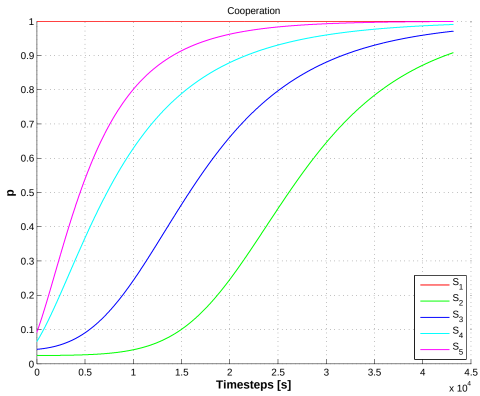
(a) (b)

Figure 6.13: Cost (a) and Payoff (b) for $p = 1$, storage cost $= 1$

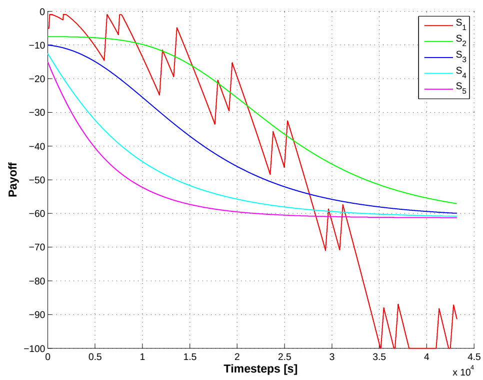


(a) (b)

Figure 6.14: Memory (a) and throughput (b) evolution over time



(a)



(b)

Figure 6.15: Cooperation (a) and Payoff (b) evolution over time

in real satellites would result in more exploitable contact windows). Furthermore as store data is an operation that requires less energy that the transmission, every satellites uses less power compared to the previous case.

6.4 Conclusions

The simplified inter-satellite connection model provides the spacecraft system a minimum autonomous capacity; data are exchanged without need for specific commands from the user and the distribution of the information among all the available elements reduces the chances that a single node runs out of memory or available datarate. During the test simulation, the hub element total memory is almost complete; it has been caused forcing satellites to have continuously operating payloads whereas communication and storage devices are design according to the payloads real duty cycles, resulting in data handling under-sizing to highlight system capacity in operating (for a limited time) even in out of nominal conditions.

CHAPTER 7

Optimisation

Satellite design is interested by countless trades between system performance, system cost, and each of the design parameters, both individually and in combination with other design parameters. This problem is exacerbated for fractionated satellites, both due to the enlarged number of design variables and due to their mutual influence on different satellites. As a result, the final design could be inefficient, leaving room for improvements in performance and reductions in lifecycle cost. Thus, a method is needed to enable a greater search of the trade space and explore design options that might not otherwise be considered during the conceptual design phase. Optimisation is one such method: by definition it is the process of achieving the most favourable system condition on the basis of a metric or set of metrics [boo98]. Within the past sixty years, different optimisation techniques have been applied to numerous complex problems, ranging from the design of airline flight networks that maximise revenues [SSL99] under scheduling constraints [Mat97] to the allocation of assets in financial portfolios [Ste99] under capital, regulatory, and risk constraints.
In its pure definition, optimisation refers to finding the absolute best solution to a problem. However from an engineering point of view this could be hardly achievable: due to the dimension of the search space, the non-linearities in the design process and the combinations of continuous and discrete variables, conceptual design often results in NP-hard problems, thus requiring considerable resources in order to be completely solved. Furthermore design problems tend to be combinatorial in nature with discrete variables having nonlinear relationships, so classical optimisation techniques that require continuously differentiable convex functions, like the simplex method, gradient descent and quasi-newton methods, cannot be used. Instead, particle (or agent) based methods does not require the optimisation problem to be differentiable therefore they

are suitable to be used on optimisation problems that are partially irregular, noisy or change over time [PKB07].

7.1 Particle Swarm Optimisation

7.1.1 Informal description

Particle Swarm Optimization (PSO) was first introduced by Dr. Russell C. Eberhart and Dr. James Kennedy in 1995 [KE95]; their work was essentially aimed at producing computational intelligence by exploiting simple analogues of social interaction, rather than purely individual cognitive abilities. As originally described, the PSO Algorithm is *an adaptive algorithm based on a social-psychological metaphor; a population of individuals (referred to as particles) adapts by returning stochastically toward previously successful regions* [KE01]. The PSO Algorithm shares similar characteristics to Genetic Algorithm, however, the manner in which the two algorithms traverse the search space E is fundamentally different. E is a hyperparallelepid defined as the Euclidean product of D real intervals:

$$E = \bigotimes_{d=1}^{D} [min_d, max_d] \tag{7.1}$$

In order to perform a search, PSO makes use of agents called *particles*, which move at every step (*iteration*). The set of S particles is called the *swarm*. The swarm topology defines how particles are connected, thus controlling information sharing; when a particle is informed by another one, it becomes aware of the previous best (position and fitness) of the informing particle. Directly connected particles are called *neighbours*. A particle is made of:

- $\mathbf{x}(t)$ position at time t; it has D coordinates.
- The fitness f defined on E
- $\mathbf{v}(t)$ the velocity (in fact a displacement) at time t, which will be used to compute the next position. It has D components.
- $\mathbf{p}(t-1)$ previous best position. It has D coordinates.
- $\mathbf{p}_g(t-1)$ is the previous best position found in the neighbourhood. It has D coordinates.

In PSO original implementation, the search is performed in two phases -initialisation of the swarm, and then a cycle of iterations- as briefly depicted in Alg. 8:

- Initialise a population array of particles with random positions and velocities on D dimensions in the search space.
- Loop

- For each particle, evaluate the desired fitness function in D variables.
- Compare particle fitness evaluation with its historical best $f(\mathbf{p})$. If current value is better, then set $f(\mathbf{x})$ equal to the current value, and $\mathbf{p}$ equal to the current location $\mathbf{x}$ in D-dimensional space.
- Identify the particle in the neighbourhood with the best success so far $f(\mathbf{p}_g)$, and assign its index to the variable g.
- Change the velocity and position of the particle; $U(0, \phi_i)$ represents a vector of random numbers uniformly distributed in $[0, \phi_i]$ which is randomly generated at each iteration and for each particle.
- If stop criterion is met, exit loop.

Algorithm 8 Particle Swarm Optimisation

for $i = 1$ to S **do**
  $\mathbf{p}_i = \mathbf{x}_i$
  $f(\mathbf{x}_i) = f(\mathbf{p}_i)$
end for
repeat
  for $i = 1$ to N **do**
    if $f(\mathbf{x}_i) > f(\mathbf{p}_i)$ **then**
      $\mathbf{p}_i = \mathbf{x}_i$
    end if
    $g = i$
    for $j = 1$ indexes of neighbours **do**
      if $f(\mathbf{x}_j) > f(\mathbf{p}_g)$ **then**
        $g = j$
      end if
    end for
    $\mathbf{v}_i(t) = \mathbf{v}_i(t-1) + U(0, \phi_1) \otimes (\mathbf{p}_i(t-1) - \mathbf{x}_i(t-1))$
          $+U(0, \phi_2) \otimes (\mathbf{p}_g(t-1) - \mathbf{x}_i(t-1))$
    $\mathbf{v}_i(t) \in (-\mathbf{V}_{max}, +\mathbf{V}_{max})$
    $\mathbf{x}_i(t) = \mathbf{x}_i(t-1) + \mathbf{v}_i(t)$
  end for
until stopping criteria

Algorithm behaviour is governed using a a small number of parameters that need to be fixed. One parameter is the size S of the population (often set empirically on the basis of the dimensionality and perceived difficulty of a problem, commonly ranging between 20 and 50 [SE98]). The parameters ϕ_1 and ϕ_2 control the magnitude of the random forces in the direction of personal and neighbourhood best $\mathbf{p}_i$ and $\mathbf{p}_g$ (acceleration coefficients). The resulting components $U(0, \phi_1) \otimes (\mathbf{p}_i(t-1) - \mathbf{x}_i(t-1))$ and $U(0, \phi_2) \otimes (\mathbf{p}_g(t-1) - \mathbf{x}_i(t-1))$ can be associated to attractive forces produced by springs of random stiffness pulling a particle. Changing ϕ_1 and ϕ_2 can make the PSO more or less responsive and consequently less or more stable, with particle speeds increasing without control. Speed control was initially achieved by hard-bounding velocities so that each component of $\mathbf{v}_i$ is kept within a given range $[\pm\mathbf{V}_{max}]$. The optimal value of $\mathbf{V}_{max}$ is problem-specific, but no reasonable rule of thumb is known and furthermore it influences the balance between exploration and exploitation. Successive versions of the PSO (inertia weight, constriction coefficients) have addressed the problem [PKB07]. The choice for neighbourhood topology determines which individual to use for $\mathbf{p}_g$. Several topologies have been applied to PSO:

- proximity in the search space (Euclidean neighbourhood)
- g_{best} the best neighbour in the entire population influences the target particle (can be conceptualised as a fully connected graph); it converges fast, as all the particles are attracted simultaneously to the best part of the E. The drawback is that if the global optimum is not close to the best particle found so far, swarm exploration toward other areas will be limited: the swarm can be trapped in local optima.
- l_{best} particle is affected by a restricted number of other particles; this topology had the advantage of allowing parallel search, as sub-populations could converge in diverse regions of E. The swarm will converge slower but can locate the global optimum with a greater chance.

As the l_{best} topology seemed better for exploring the search space while g_{best} converged faster, a wide range of adaptive topologies (that begin the search with an l_{best} lattice and increase the size of the neighbourhood, thus achieving fully connected population by the end of the run) have been developed [Sug99].
Different neighbourhoods can be characterised in terms of two factors: the degree of connectivity, K, that measures the number of neighbours of a particle and the amount of clustering C, that measures the number of neighbours of a particle that are also neighbours of each other.

7.2 Design Parameters

In order to work and find a solution within acceptable time constrains, PSO internal parameters as well as design variables (fractions) have to be coherently set and used.

7.2.1 PSO Parameters

Swarm size and neighbourhood topology have been chosen after a series of test runs, in order to evaluate whose parameters where more effective in solving the distribute design optimisation. According to literature [Sug99], a priori estimate of S can be derived from D as $S = 10 + [2\sqrt{D}]$; however, this parameter is problem-specific, so test cases have been analysed with 4 different S, 10, 14 (suggested value), 20 and 30.

The choice of the connection topology influences the number of iterations required to obtain a solution; various schemes have been evaluated:

- Full, $K = S$
- Ring, the neighbourhood of the particle i is $\{i - mod(S); i; i + mod(S)\}$
- Von Neumann, a two dimensional grid with neighbours to the N, E, W and S
- Adaptive random topology [Cle06]

Adaptive random topology is a variation of the stochastic star where after each unsuccessful iteration each particle informs at random K particles, thus modifying the information graph. K can vary between 1 and S with non-uniform distribution. Convergence results for the complete problem (all the 4 fractions, described in Sec. 7.2.2) are illustrated in Fig. 7.1. Full markers have been used to indicate swarm size equal to 30, empty markers have been used to depict a population of 14 agents, the proposed value from literature. Convergence of smaller (10 and 20) populations have been dropped for clarity as they were nearly superposed to the 14 elements result. Due to the stochastic contributes in the PSO algorithm, 3 runs have been made for every method. Adaptive, Von Neumann and Full topologies have similar performances,

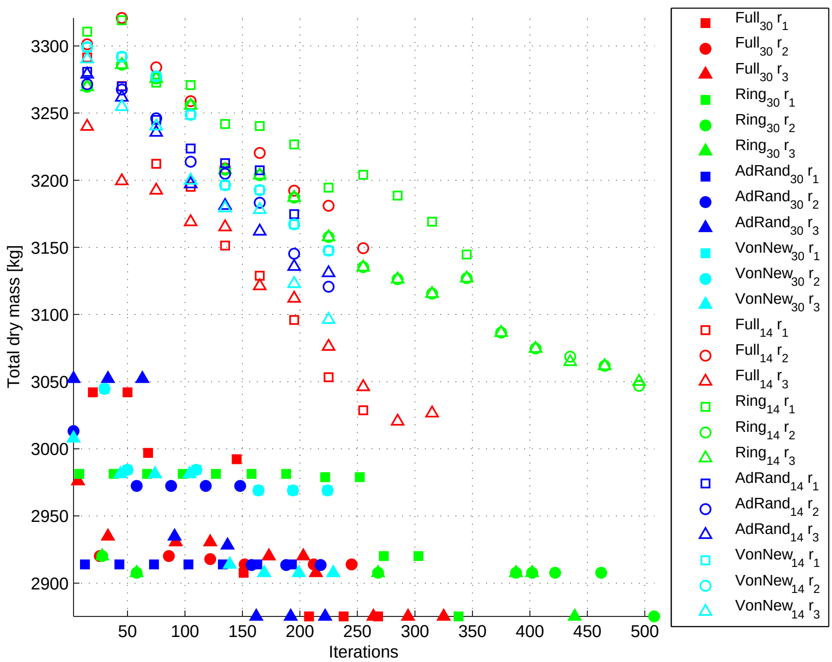


Figure 7.1: PSO convergence

whereas Ring topology required a larger number of iterations to find the optimal solution. All methods lead to equivalent engineering solutions triggered by slightly different design parameters, due to non linearities related to telecommunication and data handling subsystems, described in detail in Section 7.3.3. The largest population lead to an earlier convergence; however subsequent test with an even larger population size, 40 agents, highlighted no performances improvements.

7.2.2 Free Variables

The free design variables, that have been used as optimisation parameters, are the fractionation levels for the 4 subsystems that have been identified as technically compatible with the current state of the art, Sec. 4.2. In order to be used within the PSO algorithm, fractions have to be translated into range information: minimum and maximum allowable values will constitute the upper and lower boundaries of the state vector that is base that allows the algorithm to seed new particles. For simplicity fractionation has been expressed using values in the range ± 1 referred to the "main" or first satellite in the cluster:

- positive values indicate that the main satellite covers a *master* position (provides a remote resource, e.g. is the comms hub or the power beaming source).
- negative values are associated to *dependant* roles (the main spacecraft exploits resources shared by the other elements in the cluster).

For coherence, the other modules in the spacecraft system hold the opposite role with a reduced effectiveness, such that the sum of the parameters for a specific subsystem over the cluster is equal to zero. Fraction parameters are then mapped into minimum-maximum values to describe the amount, and not only the percentage, of resources that have to be allocated, Tab 7.1. ADS is not reported in table, as it is not associated to a divisible resource; satellites can either have a *master* role or a *dependant* but attitude determination could not be "partially shared" among two elements.

	Minimum	Maximum
EPS, power	$\{navigation, science\}_{master}$	$\sum_{i=1:N_s}^{i\neq master}\{navigation, science\}$
TT&C, DR	$\{telemetry, science\}_{master}$	$\sum_{i=1:N_s}^{i\neq master}\{telemetry, science\}$
DH, memory	$\{science\}_{master}$	$\sum_{i=1:N_s}^{i\neq master}\{science\}$

Table 7.1: Minimum and maximum shared resource amount

7.3 Fractional parameters influence

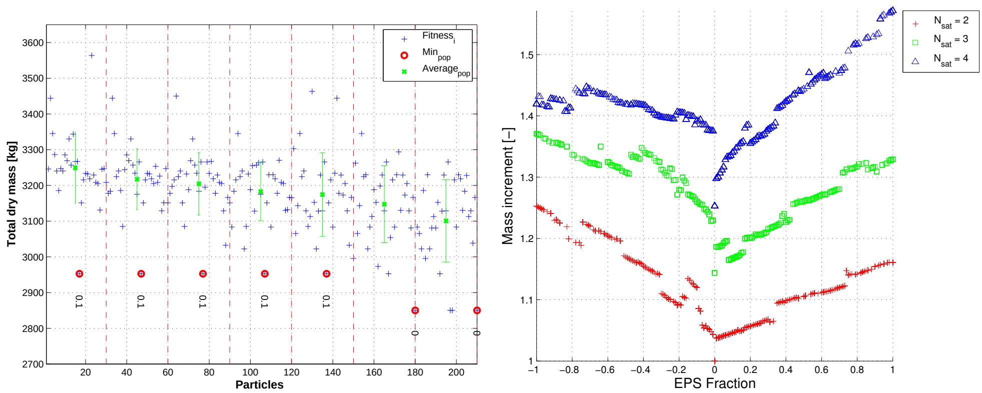
Influence of fractional parameters on optimal solution has been analysed, both individually (1 free, 3 constrained) and in conjunction (all 4 free). The Aura mission has been used as reference satellite, population size 30, fully connected topology; spacecraft system composed of 2, 3 and 4 elements have been tested. The choice of Aura has been dictated by:

- Accuracy of the SAP solution for this specific satellite
- Number (6) and variety of payloads (SARs, imager) that allows a sharing over different modules maintaining challenging requirements for all of them
- High accuracy requirements are mainly dictated by one of the payloads, the others could work with a lower resolution, thus introducing differences in the ADS

A more comprehensive description of the satellite and its mission has been stated in the appendix.

7.3.1 EPS

Convergence for EPS fraction parameter has been reported in Fig. 7.2-a and optimisation results for different configurations and cost functions has been resumed in Tab. 7.2. Variations have been evaluated with respect to the monolithic configuration. Best-case transfer efficiency of 40% has been used. Total mass is dominated by subsystems duplication causing a substantial boost; cost increase is less marked as a large part of the total cost depends on the launcher that is the same for the 3 configurations. The number of iterations has a minor dependence on the selected fitness function, and is mainly dictated by the number of satellites in the system. The mass-



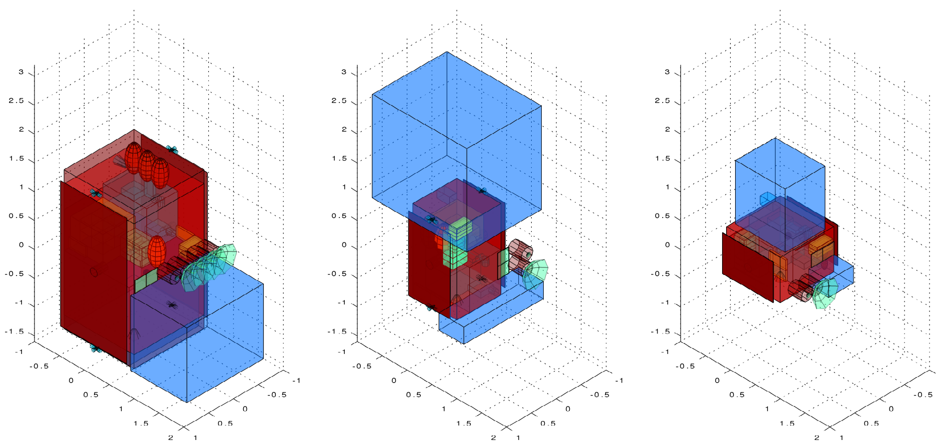
(a) (b)

Figure 7.2: Convergence (a) and mass increment (b) function of EPS fraction

EPS fraction relationship is better depicted in Fig. 7.2-b where the mass increment is reported for different configurations (number of satellites) and fraction parameter value; the chart highlights that the minimum mass (and cost) solution is associated to a non-distributed power system. Points are not symmetrical with respect to the zero value; this is due to non symmetrical map parameter-power that has to be shared. The two branches of the curve for $N = 4$ also exploit a different slope with respect to the other cases: increasing the number of elements in the spacecraft system, payloads have been re-distributed, reducing the number of those installed on the main satellite and relocating them on the remaining modules, thus increasing the amount of power that has to be produced and transmitted. A comparison between non fractioned, Fig. 7.3 and fractioned EPS configurations, Fig. 7.4, has been reported; although not shown, the surface of solar panels in the second case is nearly doubled, as so is the batteries number, thus leading (among the others variations) to a larger structure for the third satellite in Fig. 7.4.

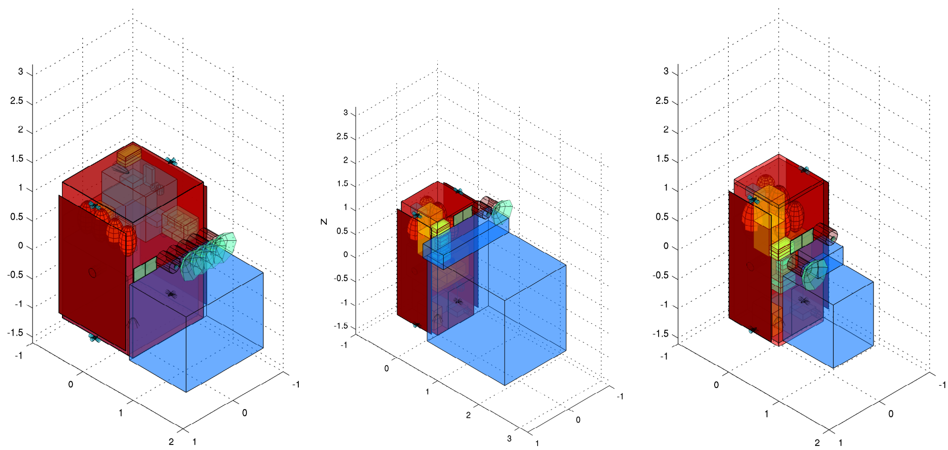
N	Estimated Mass			Estimated Cost		
	Fract. EPS	Particles	Variation	Fract. EPS	Particles	Variation
2	0.	132	+15.7%	0.	143	+8.78%
3	0.	181	+32.9%	0.	192	+15.34%
4	0.	206	+45.6%	0.	199	+20.92%

Table 7.2: Optimisation result for EPS fractionation



(a) (b) (c)

Figure 7.3: Configuration for $N = 3$, main satellite (a) and two dependant elements (b & c), distributed payloads but no power beaming



(a) (b) (c)

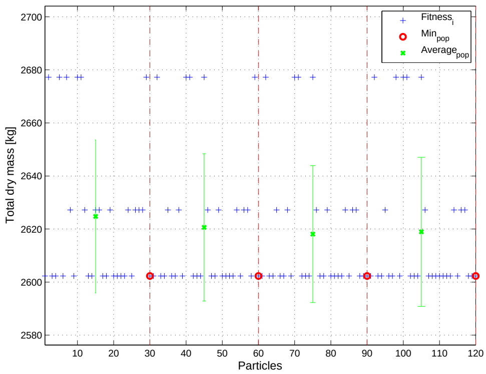
Figure 7.4: Configuration for $N = 3$, main satellite (a) and two dependant elements (b & c), distributed payloads and EPS fraction parameter equal to -.3

7.3.2 ADS

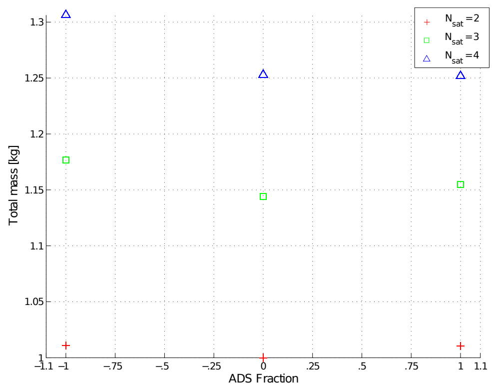
ADS parameter is allowed to assume only 3 integer values that reflect the 3 possible solutions to remote attitude determination *master*, *slave* or distributed functions (in other words the conventional solution), thus a branch-and-bound type optimisation algorithm [LD60, JKDW01] would have been a more fitting approach. Instead, the continuous values generated within the PSO update step have been restricted to integers, Fig. 7.5-b; this kind of constrains on the position and velocity of the particles is technically possible [MKS$^+$08, PV02] but in some cases could obstacle the convergence. However, due to the extreme simplicity of the problem (only 3 possible values) the solution is found within the first population (a considerably reduced population size could have been exploited), Tab 7.3. The assemble satellites are the same found during the EPS optimisation as even in this case the monolithic solution has both mass and cost advantages over the fractionated alternatives, Fig. 7.5-b.

N	Estimated Mass			Estimated Cost		
	Fract. ADS	Particles	Variation	Fract. ADS	Particles	Variation
2	0.	30	+15.7%	0.	30	+8.78%
3	0.	30	+32.9%	0.	30	+15.34%
4	0.	30	+45.6%	0.	30	+20.92%

Table 7.3: Optimisation result for ADS fractionation



(a)

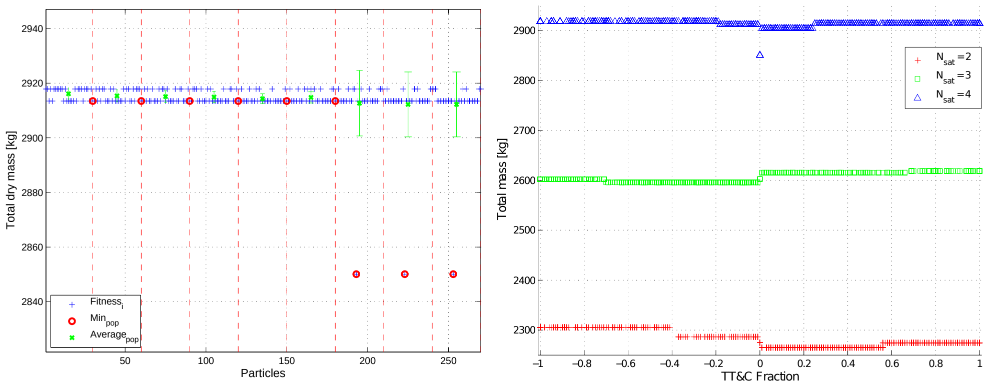


(b)

Figure 7.5: Convergence (a) and mass increment (b) function of ADS fraction

7.3.3 TT&C

Communication and data handling subsystems have been analysed separately; unlike EPS, where a clear trend is identifiable, TT&C relationship with total mass (or cost) is complicated by the presence of large areas where different fractions still lead to the same result, Fig. 7.6-b. The reason behind this undesirable phenomenon, is the "granularity" or discretisation of the components used in the database that is the base for the subsystem assembly. Each component covers a certain range of possible datarate; the requirements increment or reduction associated to fractionation could (or not) trigger a re-design. This lack of resolution causes the discontinuities as well as the incapacity to individuate a clear connection between the amount of shared resource and the spacecraft system performance. Furthermore the convergence is delayed as the PSO is not able to find a value that correspond to the desired minimum, instead a set of possible values attract the particles, Fig 7.6-a and Tab. 7.4. Communication fraction, when implemented, lead to spacecraft system whose total mass and cost could be lower than a non-resource shared configuration with the same number of satellites: cost and mass variations for $N = 2$ and $N = 3$ suggest that, when multiple satellites have to be deployed [WHMK12, GWHR12], address their communication system as a Wi-Fi network could allow a marginal improvement. However, the implication due to the software development to make the satellites able to autonomously handle to communication, may cause this advantage to be lost. Ranges corresponding to different best solutions could vary according to initial (payload imposed) and modified (fractions) requirements. One of the possible achievable configurations is depicted in Fig. 7.7.

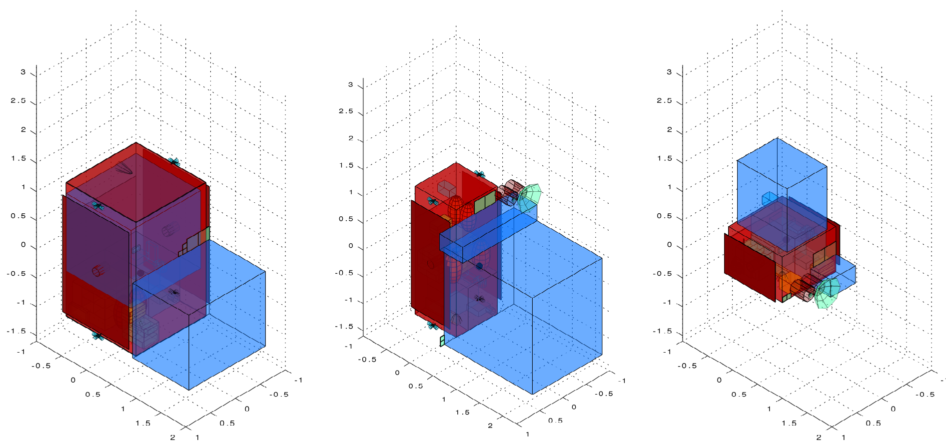


(a) (b)

Figure 7.6: Convergence (a) and mass increment (b) function of TT&C fraction

N	Estimated Mass			Estimated Cost		
	Fract. TT&C	Particles	Variation	Fract. TT&C	Particles	Variation
2	0.01÷.55	188	+15.2%	0.01÷.45	176	+8.5%
3	-.69÷-0.01	195	+32.64%	-.61÷-0.01	201	+15.28%
4	0.	210	+45.6%	0.	210	+20.92%

Table 7.4: Optimisation result for TT&C fractionation

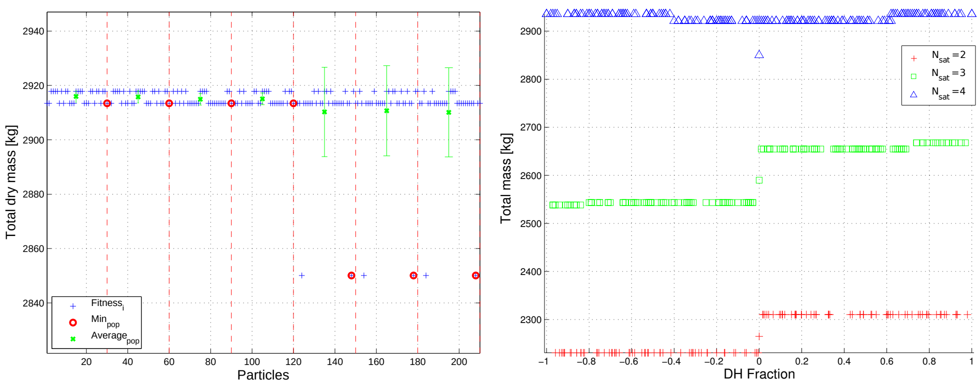


(a) (b) (c)

Figure 7.7: Configuration for $N = 3$ TT&C fraction = -.6, main satellite (a) and two dependant elements (b & c)

7.3.4 DH

DH distribution is affected by the same problem highlighted with the communication subsystem; the lack of resolution in the components database resulted in a limited number of possible solutions, each covering a part of the requirements domain. Consequently, instead of a punctual value of fractionation associated to a local minimum, ranges have been identified, Tab. 7.5. Remote data handling has been analysed without the related distributed communication system; the fractionation of the computer system gives a minimum advantage over the distribute configuration for negative fraction parameters. This causes the secondary satellite(s) to share part of the burden of the main element, resulting in smaller computational and storage units instead of a larger, and more expensive central core. For $N = 4$ the advantage is lost to excessive subsystems duplication.



(a) (b)

Figure 7.8: Convergence (a) and mass increment (b) function of DH fraction

N	Estimated Mass			Estimated Cost		
	Fract. DH	Particles	Variation	Fract. DH	Particles	Variation
2	-1÷0.01	164	+15.5%	-1÷0.01	177	+8.61%
3	-1÷-0.82	187	+32.78%	-1÷-0.82	179	+15.3%
4	0.	203	+45.6%	0.	199	+20.92%

Table 7.5: Optimisation result for DH fractionation

7.4 All fractions optimisation

The design optimisation has been applied to the Aura test satellite leaving the 4 fractioned parameters as free variables, thus allowing the algorithm to search for the combination of parameters that satisfies at best the fitness function. Estimated total cost has been used as optimum indicator. Results achieved for imposed number of satellite in the cluster are hereby resumed in Tab. 7.6 where both single to fractioned, $\Delta(Fract - Mono)$, where fitness value for monolithic and fractioned configurations are compared, and distributed to fractionated $\Delta(Fract - NonFract)$ comparisons have been reported. The parameters for used fractioned configurations have been calculated using the optimisation process; the costs for the two configurations for N = 4 are the same because the optimisation response was to prefer a non fractioned architecture, Tab. 7.8. The configurations of the spacecraft systems for increasingly larger number of elements are presented in Fig. 7.10, 7.12 and 7.14. The fractionation effects at first sight can be guessed from the presence of the parabolic antennas, required to establish downlinks with the ground stations, whereas larger beam-width patch antennas are used for local networking. Internal changes, mainly related to data handling separation are less visible. The corresponding PSO iterations, as well

N	Estimated Cost [M€]		Variation	
	Non Fract.	Fract.	$\Delta(Fract - NonFract)$	$\Delta(Fract - Mono)$
1	456.711	-	-	-
2	496.801	495.230	-0.317%	+8.43%
3	526.795	525.791	-0.191%	+15.12%
4	552.262	552.262	0.%	+20.92%

Table 7.6: Optimisation results comparison

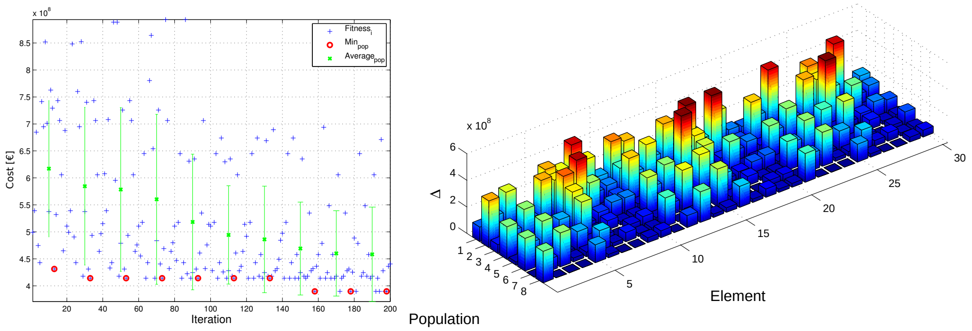
as the evolution of the state vector are reported in Fig. 7.9, 7.11 and 7.13.

Although most of the considerations about the achieved results have been

EPS	ADS	TT&C	DH	Iter.
0.01	0	-0.53	-0.24	13
0.01	0	0.55	-0.24	53
0.01	0	0.37	-1	73
0.01	0	0.61	-1	93
0.01	0	0.37	-0.24	113
0.01	0	0.37	-0.24	133
0	0	0.58	-1	178
0	0	0.38	-1	198

EPS	ADS	TT&C	DH	Iter.
0.02	0	0.88	-0.83	8
0.02	0	-1	0.03	38
-0.01	0	-0.34	0.03	68
0	0	-1	0.45	96
0	0	-1	0.45	116
0	0	-1	-0.03	136
0	0	-1	0.45	156

Table 7.7: Optimisation, parameters and iterations, $N = 2$ (left) and $N = 3$ (right)



(a) (b)

Figure 7.9: Convergence for Aura, $N = 2$, all fraction parameters

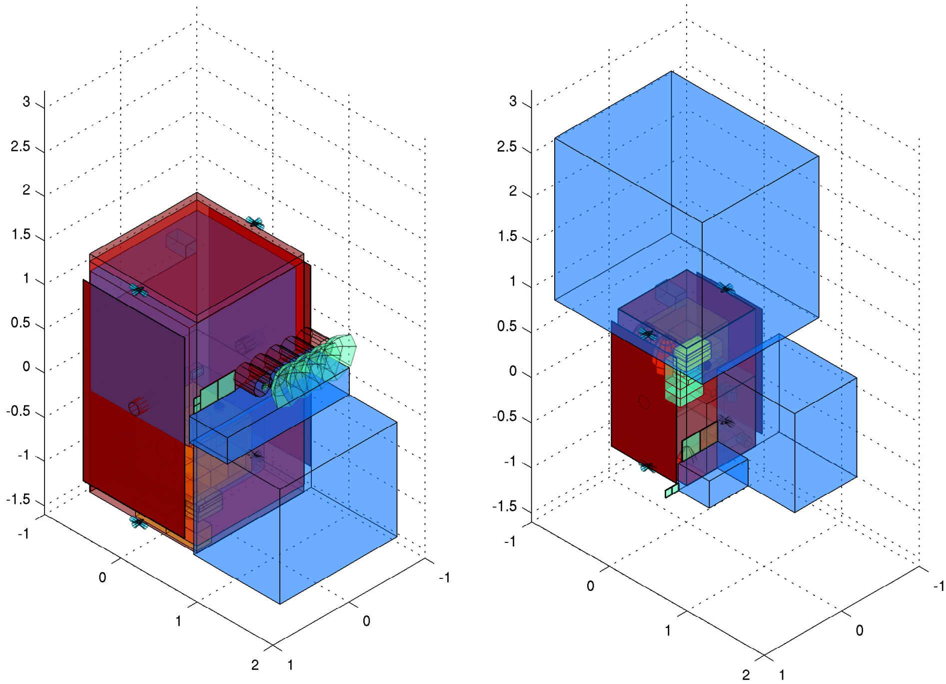
EPS	ADS	TT&C	OBDH	Iteration
0.15	0	-0.19	-1	23
0.13	0	-0.05	-1	48
0.01	0	0.15	-1	68
0.01	0	0.15	-1	88
0.01	0	0.0	-1	113
0.06	0	0.0	-0.2	135
0.01	0	0.0	0.0	164
0	0	0.0	0.0	198
0	0	0.0	0.0	218

Table 7.8: Optimisation, parameters and iterations, $N = 4$

included in the following chapter it is possible to observe that EPS and ADS fractioned solutions are never within the sets of optimal solutions whereas, at least in some cases, remote data handling and communications could allow for a (limited) cost reduction with respect to the distributed configuration. However the monolithic approach is by far cheaper, thus making the introduction of fractionation an interesting solution only if the number of satellites is larger than one due to scientific or operative motivations.

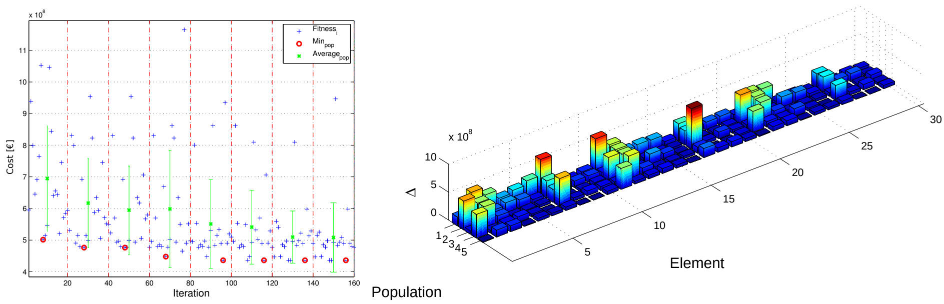
7.5 Extended design life

One of the suggested advantages of the fractioned architecture, implies that service element in the cluster during their lifetime could be either re-used to support different missions or, within the extension of the initial mission, providing a stable frame to ensure the operability of upgraded versions of the payload, thus reducing the overall cost as only a smaller part of the original spacecraft has to be replaced instead of the whole assembly [BLSE07]. A possible opposition to this thesis could rise from the



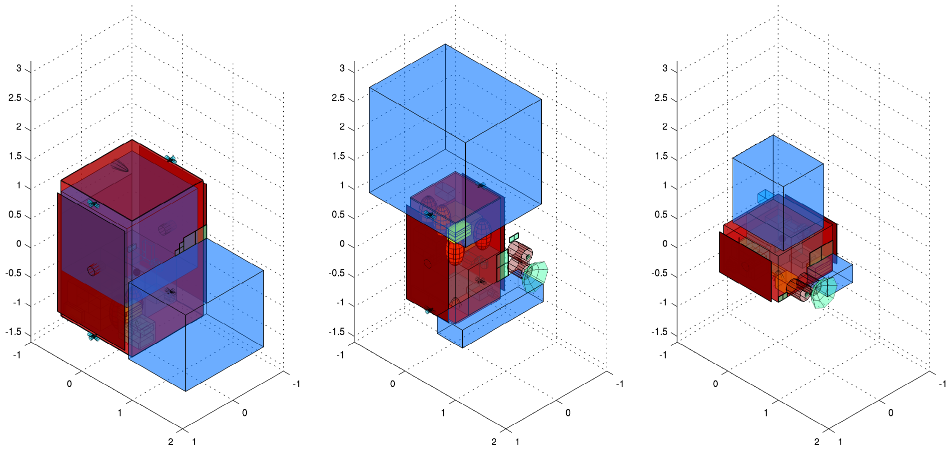
(a) (b)

Figure 7.10: Configuration for $N = 2$ TT&C fraction = -.38, DH = -1



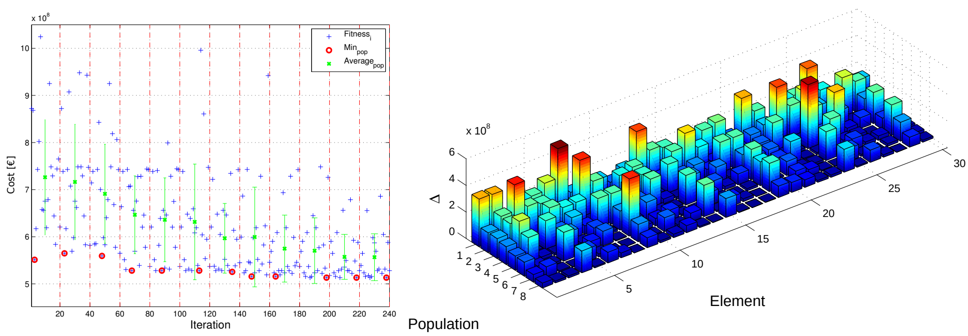
(a) (b)

Figure 7.11: Convergence for $N = 3$, all fraction parameters



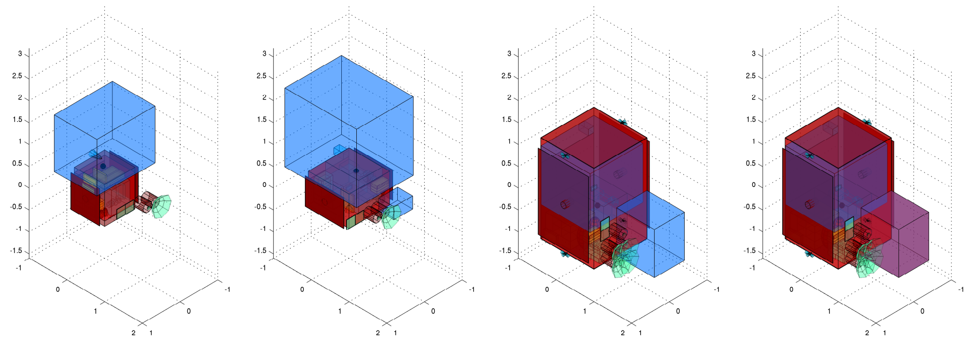
(a) (b) (c)

Figure 7.12: Configuration for $N = 3$ TT&C fraction = -1, DH = -.45



(a) (b)

Figure 7.13: Convergence for $N = 4$, all fraction parameters



(a) (b) (c) (d)

Figure 7.14: Configuration for $N = 4$ TT&C fraction = 0., DH = 0.

study of Earth-observation satellites orbital parameters: although similar, as most of them exploit LEO or sun-synchronous orbits, excluding the A-train example there are no other cases of satellites built for different observations that share the same orbit. Even minimum differences in semi-axis or (and especially) in inclination would result in large ΔV (hundreds of meters), leading to service module with modified design in order to carry since the beginning of their commissioning the propellant required to perform not only corrections but consistent orbital manoeuvres [VAl04]. Even not considering orbital changes, the influence of time, and consequently the higher reliability required, would affect the spacecraft system cost. The Aura mission has been used as test case to investigate the implication of the fractionated concept when applied to long-running observation missions. The original design life for the test satellite was 6-years; both conventional and fractionated solutions have been used to provide an operative time that is twice that value. Four concepts have been investigated:

- Long-running. Convectional approach, basically the initial satellite but designed to survive and operate for 12 years.
- Conventional, dual launch. Two identical copies of the same satellite, launched with a 6 years delay.
- Fractionated, dual launch. Two set of fractionated satellites, launched with a 6 years delay.
- Long-running, fractionated. Service module(s) design to operate for 12 years, payload replaced after 6 years.

A short resume of each configuration is reported in Tab 7.9. The costs associated to each concept have been evaluated, including satellites construction, research and development, launch and ground support at fiscal year 2013. The service module of

Configuration	Element ID	Role	Launch	Decommission
Long-Running	1	Science	2013	2025
Convectional, Dual Launch	1	Science	2013	2019
	2	Science	2019	2025
Fractionated, Dual Launch	1	Science	2013	2019
	2	Service	2013	2019
	3	Science	2019	2025
	4	Service	2019	2025
Long-Running, Fractionated	1	Science	2013	2019
	2	Service	2013	2025
	3	Science	2019	2025

Table 7.9: Extended design life concepts

the fractionated spacecraft systems has been designed payload-free, in order to provide complete communication and data handling support (comms and DH fractions equal to 1) to the observation platform. When multiple satellites have had to be launched the same year, a trade off between single and dual (smaller) launchers have been conducted; single launch has proved to be the cheaper option. The estimated costs have been reported in Tab. 7.10. The conventional configuration is, by far, the best solution. Although is the less flexible option, from a scientific point of view, as it is not possible to improve the instruments during its lifetime, its cost is considerably lower. The main reasons for its dominance are the lower cost of the satellite (compared with multiple elements configurations) and the exploit of a single, medium launcher. The long-running, fractionated architecture involves a considerably larger system cost; those are related to the need for multiple launches and the 3 satellites including one that has to remain in orbit for the whole lifetime. Possible cost reductions due to

Configuration	Total Satellites	Launches	Cost [M€]
Long-Running	1	1	772
Convectional, Dual Launch	2	2	913
Fractionated, Dual Launch	4	2	990
Long-Running, Fractionated	3	2	1075

Table 7.10: Extended design life, results

personnel training and scale factor due the production of satellites with similar design and the same base components have a limited effect. Another possible source for funds saving is the exploit if piggy-back launches for the Science elements in the fractioned configurations; however is hard to evaluate the impact of this strategy as a suitable main contractor has to be found and modifications on the satellites have to be made.

CHAPTER 8

Conclusions

The analysis of the design optimisation applied to the Aura satellite using both the number of satellites and the fractioned parameters as free variables has allowed some general observations about the influence and effect of the fractioned concept:

- EPS and ADS fractions are rapidly discarded from the candidate solutions; within the first few populations both parameters are set to 0 or a really close value. This confirms the output of the single parameter optimisations.
- The PSO requires a small number of iterations to reach a candidate solution; however the non-linearities highlighted during the DH and TT&C analysis obstacle the convergence thus causing the algorithm to perform additional steps, changing the parameters without achieving improvements to the fitness function. The heuristic solution to address the problem could involve the modification of the convergence condition to prevent further evaluations when multiple particles achieve the same fitness in spite of different state vectors. Nonetheless a similar approach would cause the PSO to be entrapped by local minima [SE98]. A minimum of 3 populations with the same best fitness has been used as minimum requirement in order to stop the optimisation (value estimated from the observation of the DH and TT&C graphs).
- DH and TT&C fractions combined achieve a marginally better fitness value than non-fractioned spacecraft systems given the same number of elements; the difference is so negligible that the mentioned development costs for the sharing system are likely to frustrate the obtained advantage.
- Monolithic configuration could fit in a medium class launcher (in this case, the DeltaII-7400 was selected), whereas multiple modules require a larger rocket

mainly due to fairing dimension requirements (as the Atlas V-521); this alone introduces a nearly 40 M€ cost difference.

The traditional configuration is, with a certain margin, the optimal solution when costs are used to evaluate the performances. According to literature, [BLSE07, MW06] fractioned satellite cost over lifecycle should be on average lower than an equivalent monolithic system when failures (both satellites and launchers) are considered. Those events would result in the necessity to replace the satellite and an approach aimed to divide the spacecraft among smaller elements would facilitate the process. However those calculations have been made in 2004; the current trend in launchers reliability is largely positive (3 accidents over 81 launches in 2013 [Spab]) and the chances for a launch failure are now considerably lower [SA98, GPC05].
Another motivation related to the exploit of the fractioned architecture is that small, standard platforms would reduce development time and costs. But, under the pressure of the increasingly capable CubeSats, this is already happening and almost every satellite-producer has engineered versatile small to medium size standard satellites that could be rapidly integrated with customer-provided payloads.
Moreover the drop out of the F6 project due to, among the others, the lack of progresses in the secure networking in Space introduces the legitimate suspect that the research for a Space-qualified application of remote computing is quite far from being trivial or just a reverse-engineering of terrestrial applications.
And finally the functional decoupling raises more problems than it can solve. The main idea, divide payload requirements from "service" requirements, has to face the current technology. The problems encountered with the proposed EPS have been:

- Low efficiency of the available technologies; both energy producer and energy-fuelled spacecraft suffered mass and cost penalties caused by the extended solar arrays, advanced power distribution units, enhanced thermal control system.
- The module with the payload still has to obey requirements imposed by the EPS: solar cells on the Sun-facing panels have been replaced with monochrome, master-satellite aimed cells (or even worse a hemispherical radome to collect microwaves).
- Configuration issues introduced by the necessity to have both traditional components to ensure survival in case of master satellite failure and power-beam dedicated elements.

Similarly, the proposed fractioned attitude determination has come to naught due to sensor improvements and miniaturisation; the complications related to the exploit of visual (equivalently LIDAR or radio-frequency) sensors to evaluate the attitude of another spacecraft could be justified when a docking manoeuvre with a non-collaborating vehicle is attempted, whereas using it as standard way to evaluate the relative orientation it does not allow neither an accuracy improvement nor a significant cost/mass

reduction. Furthermore the system is not fail-safe unless a mean to ensure independent, absolute attitude determination is ensured also for slave elements (enduring a reduced accuracy). Alternatively the remote attitude determination could be a feasible solution when a single "main" satellite is also responsible for the control of a swarm (tens or more)of proximity flying nano satellites, each too small to host a complete sensor suite by its own; however the marketing of CubeSat-aimed complete ADCS platform with integrated nano star tracker [BCT13, PSH13] might have filled the niche where remote ADS could have been employed.
On the opposite, remote communication and data handling could result in favourable configurations. As already mentioned the concept is not too different from the TDRS, although reduced to a smaller scale and with limited range capability. When applied to distributed satellites, they allow a limited improvements on the total cost; satellite to satellite communications have already been used, although the missing step is the technology required to establish an -at least semi-autonomous- network able to ensure safe and reliable data exchange. A minimum of redundancy has been introduced, imposing the slave satellites to be able to receive commands and download their telemetry even in case of main satellite malfunction. Nevertheless also satellites communication are changing: thanks to US and European data relay networks, new spacecraft have unparalleled downlink capacity thanks to the possibility to take advantage of longer contact windows towards geostationary satellites exploiting extremely high datarate (exceeding 1 Gbps) laser systems [EIW+12]. This evolution would reduce the need for locally-fractioned satellites, as bridges based on "standard" data relay would provide access to higher grade performances, unless the fractioned system itself is equipped with a comparable system. The potential advantageous applications of TT&C/DH sharing are restricted to spacecraft systems that, due to technical or scientific purposes, are forcedly composed by more than one satellites. In that case, the concentration of the communication equipment on a single element, instead of its duplication on both modules allow for a reduction of the mass, cost and complexity of the resulting system.

8.1 Open points and Further developments

All of the made analysis are based on cost models that have been only partially independently verified; the expenses for research and development have been evaluated using CERs that have been tailored on conventional satellites. Furthermore, even when used within their range of applicability, CERs accuracy could be affected by mistakes, especially for new design spacecraft. Thus is possible that recurrent cost mistakes have been propagated since the beginning of this research. Both monolithic and fractioned spacecraft could have been affected. However there are no reasons to believe that (apart from the additional development cost due to still to implement networking) the errors should affect differently the two configurations: if there are faults they should produce similar effects in both cases. To this end, percentage variations have been calculated. And the test reported in Sec. 7.5 shows such large differences that even introducing inaccuracies is still unlikely that the fractioned configuration could win the direct comparison with the traditional approach. Similarly the costs associated to monolithic and multiple satellites configurations show considerable differences and only the results dictated by the distributed vs distributed-and-fractioned contrast could show a different outcome (shift in balance). Apart from a verification with another cost model, other open points that could be considered:

- Investigate if (substantial) improvements in power beaming technology could made fractioned EPS technically and economically attractive.
- Evaluate if smaller satellites would have any benefit from the fractioned architecture (the use of the PSO in this case has been postponed due to the poor performances of fractioned satellites when the original dry mass is less 1000 kg, as highlighted in Chap. 4).
- Introduce during cost evaluation the failure probability, thus retracing the speculations that originally suggested the potential benefit of the fractioned concept.
- Improve the flexibility and the reliability of the satellite design tool, as introduced in Sec. 3.4.
- Add further alternatives to used fractioned-related hardware like communication lasers instead of patch antennas.
- Implement more sophisticated control schemes both to manage the single satellite, as well as to coordinate the efforts of the complete spacecraft system.

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Aura Mission

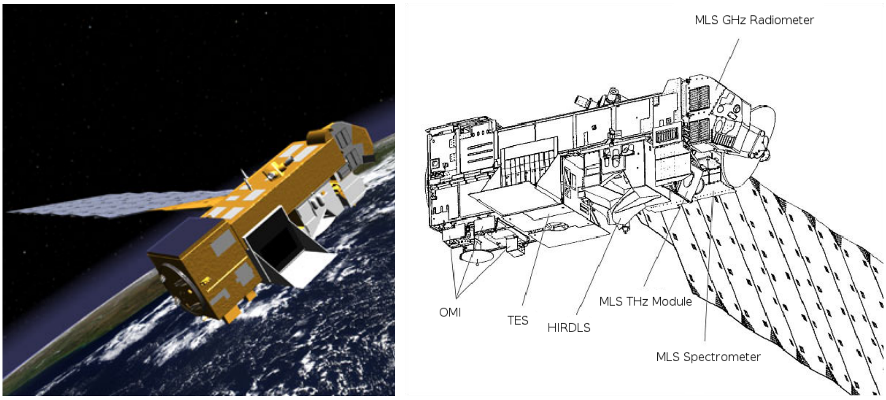
Mission

Aura (formerly EOS/Chem-1) is a multi-national NASA scientific research satellite with the overall objective to study the chemistry and dynamics of Earth's atmosphere from the ground through the mesosphere. The goal is to monitor the complex interactions of atmospheric constituents from both natural sources, such as biological activity and volcanoes, and man-made sources, such as biomass burning, are contributing to global change and effect the creation and depletion of ozone. The Aura mission will provide global surveys of several atmospheric constituents. Temperature, geopotential heights, and aerosol fields will also be mapped. In many ways, Aura is a follow-on to the very successful UARS (Upper Atmosphere Research Satellite) mission of NASA, active from 1991 to 2005. Unlike UARS, however, Aura is designed to focus on the lower stratosphere and the troposphere. Aura flies in formation about 15 minutes behind Aqua in the "A-Train" satellite constellation which consists of several satellites flying in close proximity. Each individual mission has its own science objectives; all will improve our understanding of aspects of the Earth's climate. The synergism that is expected to be gained by flying in close proximity to each other should enable the overall science results of the Afternoon Constellation to be greater than the sum of the science of each individual mission.

Spacecraft

The Aura spacecraft, like Aqua, is based on TRW's (now Northrop Grumman Space Technology) modular, standardized AB1200 bus design with common subsystems. The S/C dimensions are: 2.68 m x 2.34 m x 6.85 m (stowed) and 4.71 m x 17.03 m x 6.85 m (deployed).
Aura is three-axis stabilized, with a total mass at launch of 2,967 kg, 1,200 kg of which are scientific instruments.

The S/C design life is six years. The spacecraft structure is a lightweight 'egg-crate'



(a) (b)

Figure 1: Artist's rendering of Aura in orbit (a) and payloads disposition (b)

compartment construction made of graphite epoxy composite over honeycomb core, providing a strong but light base for the science instruments, Fig 1-a. The weight of the structure is approximately 700 kg. A single, deployable, 15 m long, flat-panel solar array with over 20.000 silicon solar cells provides 4.6 kW of power; 24 cell nickel-hydrogen battery powers the spacecraft and the instruments during the night phase of the orbit. Spacecraft attitude is maintained by stellar-inertial, and momentum wheel-based attitude controls with magnetic momentum unloading, through interaction with the magnetic field of the Earth that provide accurate pointing for the instruments. Typical pointing knowledge of the line of sight of the instruments to the Earth is on the order of one arc-minute (about 0.02°). A propulsion system of four small-thrust hydrazine mono-propellant thrusters gives the spacecraft a capability to adjust its orbit periodically to compensate for the effects of atmospheric drag, so that the orbit can be precisely controlled to maintain altitude and the assigned ground track.
Electronic components are housed on panels internally, leaving the spacecraft 'deck' available for the four science instruments, and providing them a wide field of view. The side of the spacecraft away from the Sun is devoted to thermal radiators, which radiate excess heat to space and provide the proper thermal balance for the entire spacecraft. The satellite has been launched using a Delta-2 7920 vehicle from VAFB, CA, on July 15, 2004 to the designed orbit, a Sun-synchronous circular orbit, altitude 705 km, inclination 98.7°, with a local equator crossing time of 13:45 on the ascending node. Repeat cycle of 16 days. Onboard storage capacity of 100 Gbit of payload data. The payload data are downlinked in X-band. The spacecraft can also broadcast scientific data directly to ground stations over which it is passing. The ground stations also have an S-band uplink capability for spacecraft and science instrument operations. The S-band communication subsystem also can communicate through NASA's TDRSS

synchronous satellites in order to periodically track the spacecraft, calculate the orbit precisely, and issue commands to adjust the orbit to maintain it within defined limits.

Sensor complement

The Aura instrument package provides complementary observations from the UV to the microwave region of the electromagnetic spectrum with unprecedented sensitivity and depth of coverage to the study of the Earth's atmospheric chemistry from its surface to the stratosphere. MLS is on the front of the spacecraft (the forward velocity direction) while HIRDLS, TES, and OMI are mounted on the nadir side, Fig. 1-b.

HIRDLS (High-Resolution Dynamics Limb Sounder) observes global distributions of temperature and trace gas concentrations of O_3, H_2O, CH_4, N_2O, HNO_3, NO_2, N_2O_5, CFC_{11}, CFC_{12} and $ClONO_2$, and aerosols in the upper troposphere, stratosphere, and mesosphere plus water vapour, aerosol, and cloud tops. Complete Earth coverage (including polar night) can be obtained in 12 hours.

MLS (Microwave Limb Sounder) instrument measures thermal emissions from the atmospheric limb in submillimeter and millimetre wavelength spectral bands and is intended for studies of the lower stratosphere and upper troposphere chemistry, monitoring of ozone chemistry and observation of effects of volcanoes on global change.

OMI (Ozone Monitoring Instrument) instrument is a a nadir-viewing UV/VIS imaging spectrograph which measures the solar radiation backscattered by the Earth's atmosphere and surface over the entire wavelength range from 270 to 500 nm, with a spectral resolution of about 0.5 nm. The design is of GOME heritage, flown on ERS-2, as well as of SCIAMACHY and GOMOS heritage, flown on Envisat. The overall objective is to monitor ozone and other trace gases and to monitor tropospheric pollutants worldwide. Its measurements are highly synergistic with the HIRDLS and MLS instruments on the Aura platform. The OMI observations provide mapping of ozone columns, measurement of key air quality components, distinctions between aerosol types, such as smoke, dust, and sulphates.

TES (Tropospheric Emission Spectrometer) instrument is of ATMOS (ATLAS), and AES (Airborne Emission Spectrometer) heritage and it has been developed for NASA by JPL. TES is a high-resolution infrared imaging Connes-type FTS (Fourier Transform Spectrometer), with the capability to make both limb and nadir observations. TES is a pointable instrument; it can access any target within 45° of the local vertical, or produce regional transects up to 885 km in length without any gaps in coverage. TES employs both, the natural thermal emission of the surface and atmosphere, and reflected sunlight, thereby providing day and night coverage anywhere on the globe.

Acronyms

ADCS Attitude Determination and Control System

ADS Attitude Determination and System

AuRA Autonomous Robot Architecture

biTH Bipropellant Thruster

CER Cost Estimating Relationship

C&DH Command and Data Handling

cgTH Cold Gas Thruster

COG Center Of Gravity

COTS Commercial Off The Shelf

CMG Control Moment Gyro

DH Data Handling

DR Data Rate

EDRS European Data Relay System

EIP Evaluate Input Parameters

EOL End Of Life

EPS Electrical Power System

GCE General Constrains Evaluation

GS Ground Station

hTH Hydrazine Thruster

IL Iteration Loop

IMU Inertial Measurement Unit

LEO Low Earth Orbit

MDO Multidisciplinary Design Optimization

MSAP Multiple Spacecrafts Assembly Procedure

MT Magnetorquer

NE Nash Equilibrium

PO Payoff

PSO Particle Swarm Optimization

RS Recipient Satellite

RW Reaction Wheel

SAP Satellite Assembly Procedure

SE System Evaluation

SS Source Satellite

ST Star Tracker

TCS Thermal Control System

TDRS Tracking and Data Relay Satellite

TRL Technology Readiness Level

TT&C Telemetry Tracking and Command

VPS Visual Positioning System

WCC Worst Case Cold

WCH Worst Case Hot

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A titolo di assunzione di colpa ci tengo a precisare che tutte le persone citate in questa pagina hanno svolto un ruolo nella stesura della tesi, ma ogni errore o imprecisione è imputabile soltanto a me (sono vagamente consapevole che *excusatio non petita, accusatio manifesta* ma mi sembrava doveroso specificarlo).
La voglia (di scrivere) è finita ed è tardi perché come mio solito ho rimandato fino all'ultimo, ma LaTeX mi dice che c'è un badbox se lascio il capitolo semivuoto. Tanto per la cronaca direi che l'inglese in questa sezione può essere tranquillamente abolito, come qualcuno aveva probabilmente già intuito.
I pochi (alias praticamente nessuno) che erano andati a vedere il corrispondente capitolo della tesi di laurea avranno notato che questa volta mi sto impegnando seriamente, visto che all'epoca avevo chiuso la faccenda mettendo un elenco puntato. I nomi sono più o meno gli stessi quindi ero tentato di mettere la tesi della magistrale come reference e aggiungere i nuovi ingressi.
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Riccardo

P.S. La vera citazione (che sarebbe stato brutto mettere a pagina due) che mette nella giusta prospettiva 3 (dico tre...) lauree in ingegneria:
Engineering: where the noble, semi-skilled laborers execute the vision of those who think and dream. Hello, Oompa Loompas of science!

Sheldon Lee Cooper, Ph.D.

Summary

Design, Simulation, Management and Control of a Cooperative, Distributed, Earth-Observation Satellite System

Riccardo Lombardi

The research presented in this thesis explores the potential of, and develops a framework for, the application of fractionated satellite systems to science-dedicated Earth observation missions. The label fractioned satellites highlights the physical distribution of the functionalities of the spacecraft (e.g. power generation, telecommunication, etc.) over a cluster of orbiting elements. The resultant distributed system can be seen as a free-flying payload supported by free-flying service modules. In general, the paradigm shift towards using a multiple-satellite cluster has been fuelled by the perceived advantages of increased robustness, greater flexibility, and in order to accomplish the large-scale geometries imposed by specific science objectives. Small distributed spacecraft could also guarantee better coverage than monolithic with almost comparable performances due to sensors miniaturisation. There are many ways to implement the fractionation; by interpreting literally the idea, it is possible to de-couple entirely the subsystems using different modules thus creating a completely heterogeneous system. In order to cut down the costs it seems reasonable to produce standard buses for every subsystems. Nonetheless a complete functional decomposition with the current technology not only is impractical, it is physically impossible. Every module must be able to provide by itself to basic functionality like power distribution or thermal control as well as structural integrity. Thus the most logic configuration for a fractionated spacecraft is a combination of shared resources and module-owned properties. Apart from design issue, operation phase poses a new class of challenges by itself: the introduction of the fractionated approach requires a new methodology to control and coordinate the spacecraft system efforts in order to guarantee that remote resources will be gathered and distributed according to the satellites needs; furthermore the proposed concept has been thought to be scalable to

large systems, possibly involving tenth of different elements. The operational costs of monitoring and commanding a large fleet of close-orbiting satellites is likely to be unreasonable unless the on-board software is sufficiently autonomous, robust, and re-configurable. As the goal of this research is to develop a methodology to design and simulate distributed satellite systems but no satellites with fractioned architecture actually exist, frameworks tailored to address this unique concept have to be developed and, in order to compare the performances of the fractionated system with the corresponding monolithic satellite, objective quantities have to be evaluated, like the total cost of the spacecraft including estimated development and research, ground support, construction, integration and launch.

For the design phase, several topics have been investigated, ranging from automated satellite design, analysis and evaluation of distributed resources, optimisation techniques. The first step has been the creation and validation of a monolithic-satellite aimed design tool able to assemble science-dedicated LEO satellite with a reasonable degree of confidence given informations about payload, mission and additional constraints like specific launchers or ground stations. Fundamental requirements for the tool are the capacities to provide subsystems power and mass budgets, main components list and reliabilities. A validation campaign using existing satellites as reference has been conducted. Then the capacity to handle fractioned resources has been introduced by requirements and hardware modifications. Considered fractioned subsystems are power generation and transfer by means of lasers, distributed communication and data processing and remote attitude determination. Finally a particle based optimisation method has been used to evaluate whose combination of number of satellites, shared resources and original requirements, derived from the mission payloads, could exhibit highest fitness values. The optimisation algorithm highlighted that only distributed communication and data handling could, in some cases, allow for a cost reduction whereas the reduced efficiency of the present day wireless energy transmission methods penalises this kind of solution. Analogously the shared attitude determination is not attractive due its complex implementation without significant performances improvement.

The operation phase is aimed at simulate the behaviour of the satellite during its orbit; due to the particular features introduced by the fractionation, it has been modelled to consider the additional effects introduced by fractionation, mainly the fact that resources and users are not necessarily co-located within the same satellite thus a strategy to enable and control the remote access to means must be provided. The capacity to replicate has been the first step; included elements in the simulation framework are orbit and attitude evolution including effects due to disturbances and controls, power subsystem, with evaluations of generated, consumed and available power; communication subsystems, including long and short range connections; thermal subsystem, able to evaluate the satellite components temperature and to control them using heaters; propulsion subsystem; attitude control system with simplified actuator models whose used power and propellant affect power and propulsion systems respectively; GNC algorithms. Considerable attention has been given to subsystems mutual influences, identified through a priory analysis. Additional features to account for multiple satellite simulations have been included, as a cooperation model for communication, relative attitude and position evaluation and the upgrade of GNC algorithm to manage several spacecraft. In particular a game theory based schema has been used to coordinate satellites efforts and share autonomously the communication resources.

Optimisation and simulated operation phase highlighted possible advantages and drawbacks of the fractionated concept: unlike the remote power transfer and attitude determination, shared communications and data handling could allow a cost reduction and performances improvement. However the already commissioned data relay systems could achieve similar objectives thus reducing the need for on-purpose communication modules and making the traditional configuration, with a certain margin, the optimal solution when costs are used to evaluate the performances. A comparison of different configurations to achieve extended design life also resulted that conventional approach takes advantage from the limited increase in launch and operation costs whereas fractioned satellites not only have to exploit multiple launchers but their construction cost is significantly influenced by research and development expenses.

Further studies will address the development of an increased accuracy cost model influenced by failure probability and the improvement of design and simulation frameworks.